Technical Report No. 143-1

ANALYTICAL REVIEW OF MILITARY HELICOPTER FLYING QUALITIES

(Final Report)

R. P. Walton
I. L. Ashkenas

SYSTEMS TECHNOLOGY, INC.
Hawthorne, California

August 1967

Under Contract No. NOw 64-0601-f to

NAVAL AIR SYSTEMS COMMAND
DEPARTMENT OF THE NAVY

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ABSTRACT

This report is directed at analysis and review of the current military helicopter flying qualities specification, MIL-H-8501A, and of the relevant published literature. The analytical approach rests primarily on servo-analysis of closed-loop piloting tasks and secondarily on open-loop response considerations, and such analysis and considerations are used as the basis for correlating and "explaining" the available data. This process delineates those flying qualities criteria (either in the specification or proposed in the literature) which are valid, and establishes bases for additional experiments in inadequately explored critical areas. The results are specifically applied to an itemized critique of MIL-H-8501A (excluding autoration, miscellaneous, instrument flight, and vibration characteristics).

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SYMBOLS

A	Polynomial coefficient
b	Airplane wing span
B	Polynomial coefficient
C	Polynomial coefficient
D	Polynomial coefficient
E	Polynomial coefficient
g	Acceleration due to gravity
G(s)	Loop transfer function
h	Altitude perturbation
h_1	Altitude response in one second
I_x, I_y, I_z	Moments of inertia about x, y, and z axes
I_{xz}	Product of inertia
j	$\sqrt{-1}$
K	Transfer function low frequency gain
L	Integral scale of turbulence
L	Rolling moment/I_x
L_p	$\partial L/\partial p$
L_r	$\partial L/\partial r$
L_v	$\partial L/\partial v$
L_δ	$\partial L/\partial \delta$
L'_λ	$\dfrac{L_\lambda + \dfrac{I_{xz}}{I_x} N_\lambda}{1 - \dfrac{I_{xz}^2}{I_x I_z}}$ where λ refers to any motion or input quantity
m	Mass of the aircraft

M	Pitching moment/I_y
M_q	$\partial M/\partial q$
M_u	$\partial M/\partial u$
M_w	$\partial M/\partial w$
$M_{\dot{w}}$	$\partial M/\partial \dot{w}$
M_δ	$\partial M/\partial \delta$
N	Yawing moment/I_z
N_p	$\partial N/\partial p$
N_r	$\partial N/\partial r$
N_v	$\partial N/\partial v$
N_δ	$\partial N/\partial \delta$
N'_λ	$\dfrac{N_\lambda + \dfrac{I_{xz}}{I_z} L_\lambda}{1 - \dfrac{I_{xz}^2}{I_x I_z}}$, where λ refers to any motion or input quantity
$N(s)$	Transfer function numerator
p	Roll rate
POR or PR	Pilot's rating
q	Pitch rate
r	Yaw rate
R	Rotor radius
s	Laplace operator, $s = \sigma + j\omega$
t	Time
T	Thrust
T_I	Human pilot lag time constant
T_L	Human pilot lead time constant
T_N	Human pilot neuromuscular time constant
T_λ	Time constant of λ zero or pole

u	Perturbation velocity along x-axis
u_g	Gust velocity along x-axis
U_o	Steady state velocity along x-axis
v	Perturbation velocity along y-axis
V_{as}	Steady state airspeed
w	Perturbation velocity along z-axis
W	Gross weight
x	Horizontal displacement in direction of x-axis
X	Force in x-direction divided by aircraft mass
X_q	$\partial X/\partial q$
X_u	$\partial X/\partial u$
X_δ	$\partial X/\partial \delta$
y	Side displacement in direction of y-axis
Y	Force in y-direction divided by aircraft mass
Y_p	$\partial Y/\partial p$
Y_r	$\partial Y/\partial r$
Y_v	$\partial Y/\partial v$
Y_δ	$\partial Y/\partial \delta$
$Y(s)$	Transfer function
$Y_{P(\lambda)}$	Human pilot transfer function, particularized with respect to general variable
Z	Force in z-direction divided by aircraft mass
Z_q	$\partial Z/\partial q$
Z_u	$\partial Z/\partial u$
Z_w	$\partial Z/\partial w$
Z_δ	$\partial Z/\partial \delta$
α	Angle of attack
β	Sideslip angle of attack, $\doteq \frac{v}{U_o}$

γ	Flight path
γ_1	Lock number
δ	Control deflection (Positive: forward stick, right stick, right pedal)
ϵ_λ	Pilot's error, particularized with respect to general variable
Δ	Transfer function denominator
$\Delta\lambda$	Incremental change in λ
ζ_λ	Damping ratio of λ zero or pole
θ	Pitch angle
θ_o	Angle between (untwisted) blade no-lift chord line and plane of rotation
θ_1	Pitch angle response in one second
κ	Transfer function high frequency gain
λ	General variable (u, v, w, θ, φ, ψ, etc.)
μ	U_o/ R
σ	Real part of s
σ_λ	Root-mean-squared value of λ
τ	Time delay
τ_e	Effective pilot time delay, τ plus neuromuscular time constant
φ	Roll angle
φ_M	Phase margin
φ_1	Roll angle response in one second
φ_λ	λ power spectrum
ψ	Yaw angle
ψ_1	Yaw angle response in one second
ω	Imaginary part of s
ω_c	Crossover frequency
ω_λ	Undamped natural frequency of λ zero or pole
Ω	Rotational speed (rad/sec)

Special Subscripts:

A	Roll control (Lateral cyclic pitch)
B	Pitch control (Longitudinal cyclic pitch, differential thrust, etc.)
c	Collective pitch
d	Dutch roll
o	Basic (unperturbed) condition
p	Phugoid or pilot
r	Yaw control (tail rotor or differential lateral cyclic)
R	Roll subsidence
s	Spiral
sp	Short-period
T	Thrust
λ	General variable (u, v, w, θ, φ, ψ, etc.)

Mathematical Symbols:

$<$	Less than
$>$	Greater than
$\ll$	Much less than
$\gg$	Much greater than
/	Not (e.g., $\neq$, not equal to)
$\doteq$	Approximately equal to
$\rightarrow$	Fed to; approaches
∂	Partial derivative
λ/δ	Generalized transfer function, $\lambda(s)/\delta(s)$

SECTION I

INTRODUCTION

The work reported here has been directed at conducting an analytical review of certain aspects of the current military helicopter flying qualities specification, MIL-H-8501A (Ref. 1), with the aim of laying a rational foundation for handling qualities design criteria for this class of vehicles. The review has not been confined solely to the contents of the present helicopter specification, but has been broadened to include all relevant published material; to delineate those criteria (either in the specification or proposed in the literature) which are valid; and to establish a theoretical basis for additional experimental effort where critical areas are found to be inadequately explored. The analytical approach rests primarily on servo-analysis of closed-loop piloting tasks; and secondarily on open-loop response considerations.

The need for more rational flying qualities requirements established on a combined experimental analytical basis can be seen in the much exercised controversy which exists regarding the applicability of certain empirical relationships contained in the specification. Where possible, empirical relationship based on statistical data should be kept to a minimum, e.g., many of the requirements based solely on a size effect provide little in the way of a handling qualities design guide to an individual case.

A logical step towards achieving rational criteria is the approach taken in this report wherein the helicopter requirements are based upon experimentally validated theoretical considerations. The primary effort then has been collecting, analyzing, evaluating and correlating existing handling qualities data. The evaluation effort involves the theoretical analysis of both closed- and open-loop control tasks and the correlation of these results with available data. By this process a more unified set of criteria can be evolved which can "explain" trends in the handling qualities data and predict significant effects.

Section II introduces the dynamic characteristics of the controlled element (the helicopter) along with equations of motion which are sufficient for most of the closed-loop and open-loop analyses in the report. The ranges of the stability derivatives are given for the tandem and single rotor configuration as a function of speed. Along with the aircraft characteristics the describing function of the pilot is discussed in terms of his performance and adjustment rules in order to complete the overall picture of the pilot-vehicle control loop. An example of closed-loop control is given for the simple well studied case where the control element is $K/s(s+1/T)$.

Section III contains the analyses of four closed-loop cases which are representative of the multiple-loop situation where a pilot in a compensatory task manipulates one input by observing the error in two outputs of the system. The cases studied include hover over a spot as well as lateral-directional control during landing approaches, and were analyzed with and without gust input disturbances. The results of these analyses are used to explain certain phenomena in experimentally derived data, and these data in turn are used to define the good and bad regions for the analytically-derived critical parameters.

Section IV presents certain of the open-loop analyses used to gain a better understanding of some of the miscellaneous maneuver tasks which can be analyzed and include control stick stability, "concave downward" requirement, roll responses, etc.

Section V interprets the specification in terms of the analytical results. This then becomes the specification critique which also includes recommendations as to further experimental effort in critical areas.

SECTION II

INDIVIDUAL CHARACTERISTICS OF HELICOPTER AND PILOT WITH INTRODUCTION TO CLOSED-LOOP TRACKING TASK

A. AIRCRAFT EQUATIONS OF MOTION AND THE STABILITY DERIVATIVE

The linear equations of motion, Ref. 2, were found sufficient to handle all the situations encountered in the specification review. In this part, the equations required for hover and forward flight are discussed and the equivalence between lateral and longitudinal hover is shown. The key point of this discussion is to show and compare a typical set of stability derivatives for the tandem and the single rotor helicopters. In the process of analyzing closed-loop control, the most critical situations become the hover and the forward flight approach. The terms which are usually small and do not contribute to the understanding of these control situations are omitted, and the equations of motion can be written:

Longitudinal

$$\begin{bmatrix} s - X_u & 0 & g \\ 0 & s - Z_w & -sU_o \\ -M_u & -M_w & s^2 - M_q s \end{bmatrix} \begin{Bmatrix} u \\ w \\ \theta \end{Bmatrix} = \begin{Bmatrix} X_\delta \\ Z_\delta \\ M_\delta \end{Bmatrix} \delta \qquad (1)$$

Lateral-Directional

$$\begin{bmatrix} s - Y_v & -g & U_o \\ -L'_v & s^2 - L'_p s & 0 \\ -N'_v & 0 & s - N'_r \end{bmatrix} \begin{Bmatrix} v \\ \varphi \\ r \end{Bmatrix} = \begin{Bmatrix} Y_\delta \\ L'_\delta \\ N'_\delta \end{Bmatrix} \delta \qquad (2)$$

For the case of hover the pitch longitudinal equations of motion simply become

$$\begin{bmatrix} s - X_u & g \\ -M_u & s^2 - M_q s \end{bmatrix} \begin{Bmatrix} u \\ \theta \end{Bmatrix} = \begin{Bmatrix} X_\delta \\ M_\delta \end{Bmatrix} \delta \qquad (3)$$

With the proper changes of symbols, Eq. 3 represents the lateral hover equations of motion in roll.

$$\begin{array}{lll} u \rightarrow v & X_u \rightarrow Y_v & X_\delta \rightarrow Y_\delta \\ \theta \rightarrow \varphi & M_u \rightarrow -L_v & M_\delta \rightarrow L_\delta \\ & M_q \rightarrow L_p & \end{array}$$

Because of this identity in form, the remainder of the hover work will use only the longitudinal terminology and symbols, but the results will apply to the lateral hover as well. From Eqs. 1 and 2 and the usual kinematic relationships, $\dot{h} = U_0\theta - w$, $\dot{\psi} \equiv r$, etc., the vertical and yaw equations at hover ($U_0 = 0$) are given by the single-degree-of-freedom expressions:

$$\frac{h}{\delta_c} = \frac{-Z\delta_c}{s(s - Z_w)} \quad \text{or} \quad \frac{\psi}{\delta_r} = \frac{N_{\delta_r}}{s(s - N_r)} \qquad (4)$$

Control situations having this transfer function form for the controlled element have been studied extensively as regards closed-loop pilot control e.g., Refs. 2–5. This same form also applies to the pitch and roll hover equations when the speed stability derivatives can be neglected ($M_u = -L_v \doteq 0$).

$$\frac{\theta}{\delta_B} = \frac{M_{\delta_B}}{s(s - M_q)} \quad \text{or} \quad \frac{\varphi}{\delta_A} = \frac{L_{\delta_A}}{s(s - L_p)} \qquad (5)$$

Although the speed stability derivatives can be rather large, they still can be neglected for purposes of closing the attitude loop only in the absence of external disturbances (see Section III-A). For orientation purposes, the ranges of derivatives are given for the single- and tandem-rotor helicopters (see Fig. 1). These derivatives represent the basic airframe without augmentation and the sketches indicate the trends for

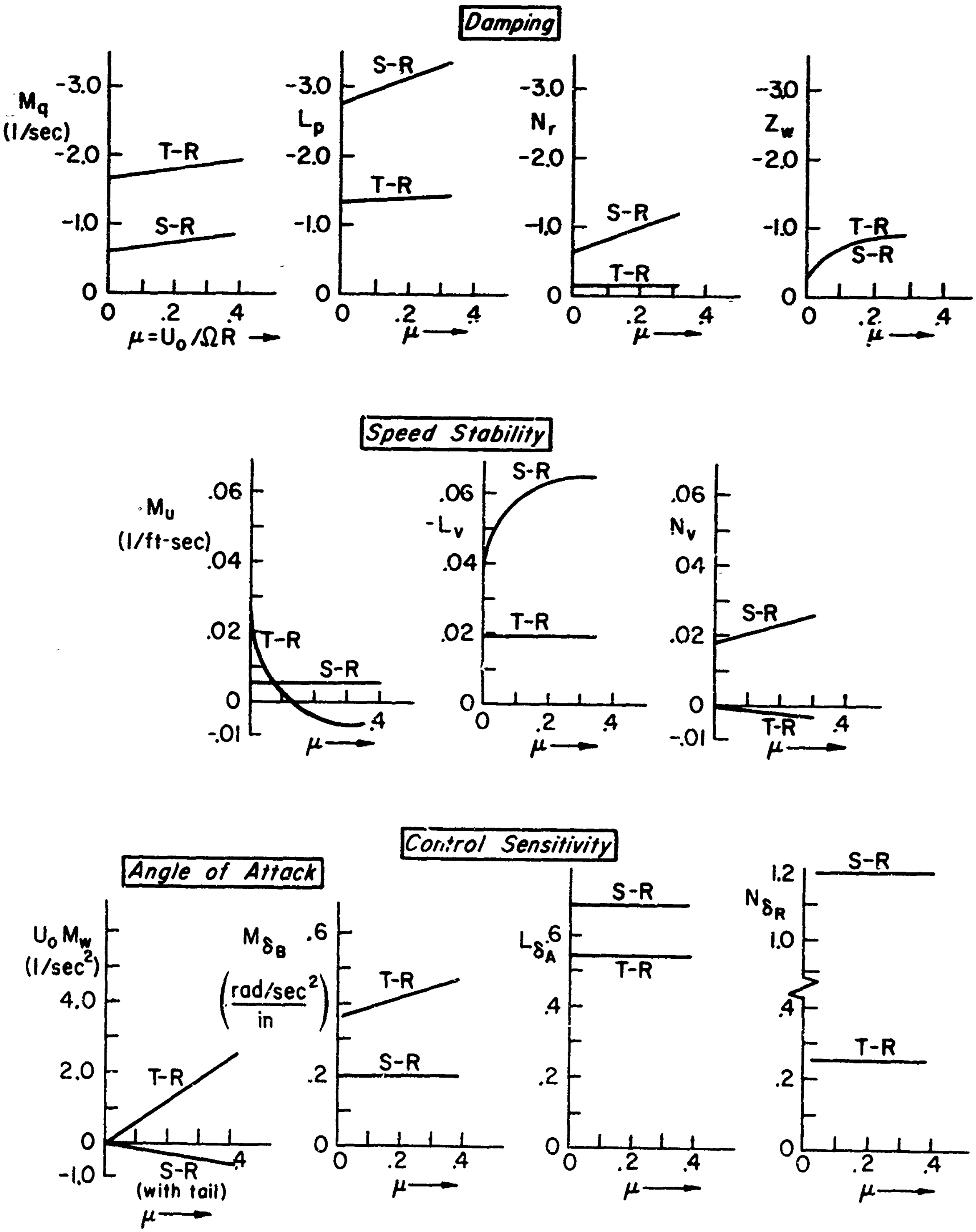

Figure 1. Variational Trends of the Stability Derivatives with Speed for the Tandem and Single Rotor Helicopter

average values with respect to each configuration. Weight or size was not found to be primary in establishing the trends even though the weights differed by a factor of five for some configurations.

Damping

A comparison of the rotary damping derivatives of Fig. 1 shows the longitudinal damping of the tandem to be greater than that of the single rotor. The situation is reversed for the lateral-directional damping where both L_p and N_r of the single rotor are greater than the tandem. While the rotary dampings are fairly constant with speed, the vertical damping, Z_w, decreases abruptly near hover and becomes approximately equal to $-.3$ (1/sec) at hover.

Speed Stability

These derivatives (M_u, L_v, and N_v) are largely responsible for the gust sensitivity about the three aircraft axes; the longitudinal derivative, M_u, is also directly proportional to the stick position stability, $d\delta_B/du$. M_u is always positive at hover, but in the case of the tandem the derivative becomes negative after transition unless certain modifications such as swash plate dihedral are made to prevent the sign change. For the single-rotor the derivative is usually small, positive, and relatively constant with speed. This indicates a stable stick variation with speed about any trim condition; however some single-rotor helicopters have a low speed stick reversal problem, possibly due to nonlinear power effects in going from one trim condition to another. For the roll and directional axes the single-rotor configuration leads to high $-L_v$ and N_v, while the tandem has a high $|L_v/N_v|$ ratio (due to the low N_v) which is associated with a large φ/β ratio. All these characteristics are essentially constant with speed except for M_u of the tandem and L_v of the single rotor near hover.

Angle of Attack Stability

As shown in Fig. 1, the pitching moment due to angle of attack is usually made stable for the single rotor by the addition of a small tail plane, while the tandem requires more sophisticated modifications

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reaction time delay and the neuromuscular lag which in this report are combined into an effective transport lag, $\tau_e = \tau + T_N$. For the purpose of constructing root locus and Bode plots, the transport lag has been incorporated into the phase and magnitude parts as

$$\varphi_\tau = -j\tau_e\omega \quad ; \quad K_\tau = e^{-\tau_e\sigma}$$

Returning to the adjustment rules, in general they can artificially be divided into two categories—adaption and optimalization. Broadly speaking, adaption is the selection by the pilot of a specific form of Eq. 6. For all cases considered in this work the closures at most require pilot lead, so the applicable form of the pilot transfer function reduces to

$$Y_p = K_p e^{-\tau_e s}(T_L s + 1) \tag{7}$$

Such adaption has been fairly well established for the controlled element forms in the present applications, and is consistent with providing good low frequency closed-loop response and absolute stability of the system. As for optimalization of the parameters K_p and T_L, they have been shown (Ref. 7) to be roughly compatible with the minimization of the rms error.

The pilot rating can be a function of many factors both of an open- and closed-loop nature, but a basic assumption of the theory presented here is that overall ratings will be most affected by flight situations requiring precise control and close attention in control modes akin to the compensatory tracking task. Consequently, as a first approximation, one can synthesize a functional relationship between pilot ratings and the primary system factors, i.e.,

$$PR = \mathrm{fn}(\sigma_\epsilon, \sigma_\delta, K_p, T_L, T_I) \tag{8}$$

The first factor, σ_ϵ, has to do with the system performance the pilot can achieve, and this is of primary importance. If performance is poor, then the remaining factors, even if favorable, will not result in a favorable rating. However, when performance in a closed-loop system is good, the pilot still wants to keep his physical output responses (if he is controlling more than one output) at some sort of a minimum; and, obviously, the smaller (within limits) his physical activity, σ_δ, the better he will

rate the pilot-airframe system. Similarly, the last two factors have to do with the pilot's mental or data-processing activity and here again the very best pilot ratings occur when his equalization requirements are minimum, i.e., $T_L = T_I = 0$ ($T_I \neq 0$ does not have an important effect on rating), and $Y_p = K_p e^{-\tau_e s}$. This "best-opinion" form of the pilot describing function is appropriate when the effective vehicle characteristics in the region of crossover have the transfer function form $Y_c \doteq K_c/s$. However, even such favorable situations will be poorly rated if the pilot's gain, K_p, is too high (as in a sluggish system) or too low (as in a sensitive system). Because the pilot attempts to keep a constant crossover frequency, ω_c, which in this case is just equal to $K_p K_c$, a K_p-induced variation of pilot rating can be demonstrated simply by varying the controlled-element gain, K_c. The result will show that an optimum gain exists, and that either increasing or decreasing this gain results in a degraded rating. The crossover frequency, ω_c, defined from the Bode plot as the highest frequency intersection of the open-loop transfer function with the zero-db line, is a good approximation to the dominant response frequency and the bandwidth of the closed-loop system. For good system performance, ω_c must be at least as large as the input disturbance bandwidth and must also come up to the pilot's expectations with regard to the dominant response time of the system. In practice the latter requirement leads to more or less constant ω_c's for given classes of tasks and simple controlled elements. For complex controlled elements the attainable ω_c depends on the total usable gain, which in turn depends on the provision of adequate gain and phase margins (about 6 db and 30°, respectively). These margins are necessary to insure adequate nominal stability and are consistent with pilot describing function measurements (extrapolated to the lower ω_c's of interest here) and with good servo practice. There is nothing really sacred about either figure, and in some instances one margin or the other may be reduced for reasons associated with a specific situation. However, in most cases analyzed here at least one margin (6 db gain or 30° phase) was maintained.

C. EXAMPLE OF CLOSED-LOOP CONSIDERATIONS FOR THE SIMPLE PITCH CLOSURE

To illustrate the use of the pilot model and at the same time to set damping requirements, consider the simple, ideal, one-degree-of-freedom pitch to longitudinal cyclic control ($\theta \rightarrow \delta_B$) transfer function:

$$\frac{\theta}{\delta_B} = \frac{M_{\delta_B}}{s(s - M_q)} = \frac{M_{\delta_B}}{s\left(s + \frac{1}{T_{sp}}\right)} \tag{9}$$

This form applies to most all hover situations (see Eqs. 4 and 5) when the speed stability derivatives are zero ($M_u = -L_v = N_v = 0$). It serves as a logical point of departure for later considering the implication of the outer-loop closures in position. Control of pitch is the primary piloting task necessary for maintaining or changing the position in hover. Regulation of the pitch angle to maintain position is a compensatory tracking task wherein the pilot applies cyclic control to change attitude as a function of his observed position error.

In exploring the closed-loop implications of ideal pitch control, the pilot's activities are characterized by his experimentally observed transfer function (Refs. 5 and 6) fitted to the simple form of Eq. 7. Accordingly, the complete open-loop transfer function in its simplest applicable form is given by Eq. 10.

$$Y_p \frac{\theta}{\delta_B} = \frac{K_p e^{-\tau_e s}(T_L s + 1)M_{\delta_B}}{s\left(s + \frac{1}{T_{sp}}\right)} \tag{10}$$

For low values of T_{sp}, corresponding to high pitch damping, the controlled element dynamics, $Y_c \equiv \theta/\delta_B$, approach the simple K_c/s form (i.e., in Eq. 9 for $T_{sp} \rightarrow 0$, $\theta/\delta_B \rightarrow M_{\delta_B}/s$). Under these conditions pilot lead is unnecessary for good closure, i.e., $Y_pY_c \doteq K_pK_c e^{-\tau_e s}/s$, and the only pilot adaption requires is on the value of gain, K_p. For this simplest of all closed loops, the open-loop gain determines the gain-crossover frequency, ω_c; i.e., $K_pK_c = \omega_c$. The corresponding phase margin is given by

$$180^\circ - \measuredangle Y_pY_c = 90^\circ - 57.3\,(\tau_e\omega_c) \tag{11}$$

Since the required pilot adaption is a minimum, pilot opinion of the K_c/s-like controlled element is invariably good provided the value of K_p is in some optimum region, <u>and</u> provided the system bandwidth, determined by ω_c, is greater than the input disturbance bandwidth. Both of these auxiliary requirements demand some knowledge of the possible or likely values of ω_c. There are a number of measurements of varying validity (Refs. 5 and 6) which indicate that, for single-loop attitude closures, values of ω_c are $2 \pm 1/2$ rad/sec for K_c/s controlled elements; and these values are essentially constant with varying input bandwidths provided these bandwidths are less than ω_c.

The basic data of Ref. 7 show additionally that regardless of the controlled-element form the complete open-loop describing function, Y_pY_c, can be approximated in the <u>crossover region</u> by

$$(Y_pY_c)_{\text{crossover}} \doteq \frac{K_pK_ce^{-\tau_e s}}{s} \tag{12}$$

In view of the basic desirability of K/s-like crossovers (for either automatic or manual control), the pilot's lead adaption for closed-loop control of pitch will be to effectively cancel the pitch subsidence mode; i.e., for $T_L \doteq T_{sp}$, Eq. 10 looks like Eq. 12 (see Fig. 2). Obviously, the pilot cannot generate sufficient lead to allow T_L to follow T_{sp} as it approaches infinity ($Y_c \rightarrow K_c/s^2$) but will, instead, be limited to a maximum value of T_L around 5 (Ref. 8). Also, T_L will not follow T_{sp} as it approaches zero ($Y_c \rightarrow K_c/s$) but will, instead, precede it and approach zero when the phase lag contributed by T_{sp} at ω_c becomes permissibly small (at a value of T_{sp} below about .5).

The outcome of the pilot's adaptive behavior is that the closed-loop performance is relatively insensitive to variations in T_{sp}, i.e., the pilot adapts in such a way as to effectively cancel T_{sp} and thereby makes all systems look like K_c/s. Therefore, provided ω_c is greater than ω_i and K_c is selected at its optimum gain, the pilot's opinion changes with T_{sp} is a function only of the required T_L adapted by the pilot. From

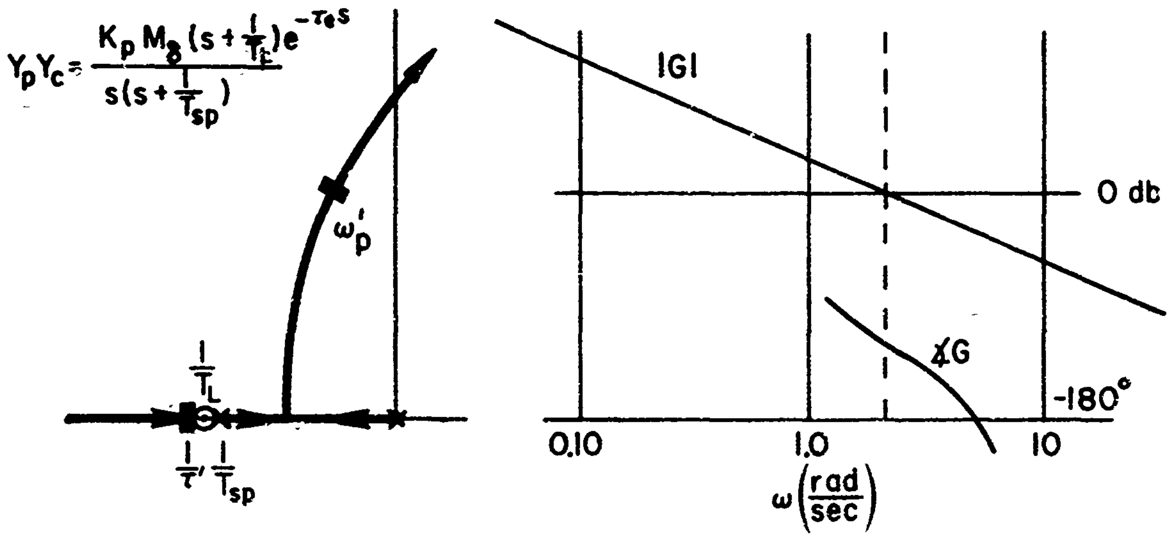


Figure 2. Sketch of Pitch Closure for $T_L \doteq T_{sp}$

pilot opinion ratings obtained with the experiments surveyed in Ref. 8 (single-loop pilot-vehicle systems with random forcing functions), an average rating increase (degraded opinion) of about three Cooper points was observed in going from best K_c/s to best K_c/s^2 (i.e., gain, K_c, for best opinion). Thus the rating increment associated with a finite value of T_{sp} will depend on the pilot-adapted value of T_L and will vary with T_{sp} roughly as shown in Fig. 3. For increasing T_{sp}, the degradation of the rating first appears at a value of T_{sp} somewhere around 1.0, the point, roughly, at which lead generation first becomes necessary. These simple tasks which employ random input forcing functions offer the most direct

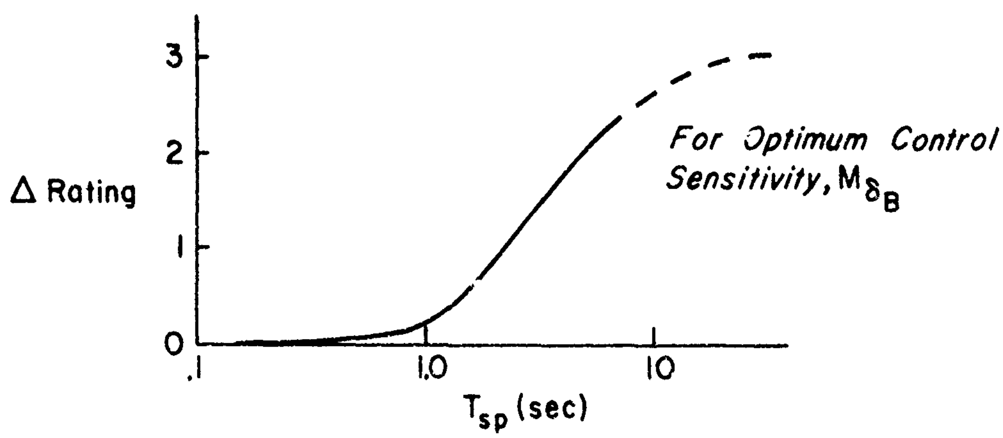


Figure 3. Rating Change as a Function of the Time Constant, T_{sp}

means of determining the valid effects of pitch damping ($-M_q = 1/T_{sp}$) on pilot rating trends for vehicles subject to external disturbances. This result has been shown in Ref. 8 to be consistent with the notion that closed-loop tracking tasks are generally more demanding as regards system dynamics than open-loop tasks. Reference 8 also presents data which show that pilot rating trends with damping are essentially the same for all axes (i.e., pitch, roll, and yaw).

For vehicles which are not exposed to external disturbances, the rating increments (and the absolute ratings) are lower than those discussed above, presumably because closed-loop regulation activities are considerably reduced. For example, the lunar lander of Ref. 47 showed ratings of about 3.5 for a pure inertial characteristic ($T_{sp} \to \infty$) and about 2.0 for a value of $T_{sp} \doteq 0.8$. The corresponding absolute ratings for similar dynamics subject to external disturbances are about 6.5 and 2.5, respectively (Ref. 8).

D. ANALYTICAL IMPLICATIONS OF GUST INPUTS AND MULTILOOP CONTROL

Using the pilot model, an analysis was made to determine the closed-loop system responses to gust disturbances. When the pilot's task includes flying in gusty air, he becomes concerned with the variations in position (or heading), attitude, and control deflection due to the gust disturbances. For analytical purposes it is desirable to have an input spectrum that is simple in form but which adequately represents the gust phenomena. Such a model, derived from Refs. 9 and 10, is given, and its use to obtain pilot-vehicle rms responses described, in Appendix B.

The pilot-vehicle system most often assumed in these studies is given by the block diagram of Fig. 4 and is probably one of the simplest of all multiloop systems because a single input is used for controlling two outputs. This system is classified as a single-point controller. The basic case depicted is for longitudinal control in a "hover-over-a-spot" situation, but the block diagram applies, with suitable changes in the motion variables, to lateral control in hover (i.e., substitute φ and y for θ and x); to elevator control of the flight path in forward flight (substitute h for x); and, as shown parenthetically in Fig. 4, to aileron control of heading in approach (φ and ψ for θ and x).

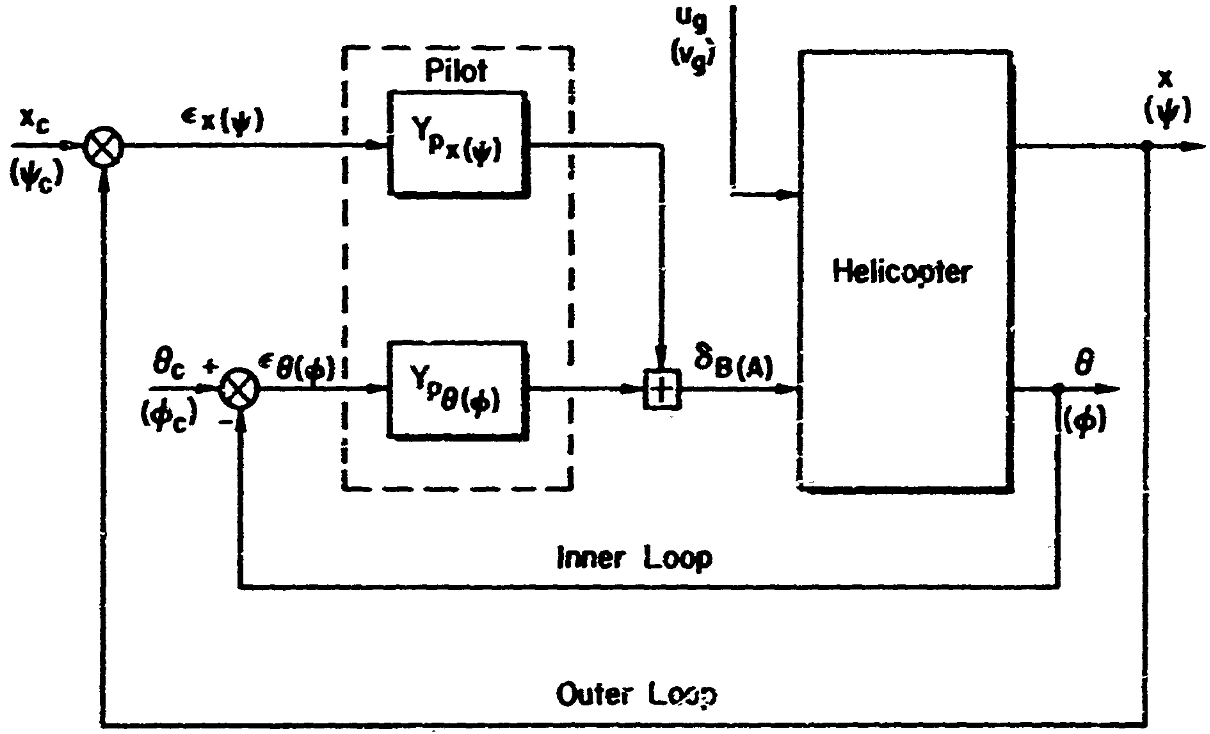


Figure 4. Block Diagram for Typical Multiple-Loop Closures of Hover Over a Spot (or Lateral-Directional Approach)

Extension of the single-loop pilot model to multiloop systems is discussed in Ref. 7 and the results obtained by the application therein can attest to the usefulness of such an extension. In the analyses the inner loop is treated first as a single-loop closure usually involving the higher frequency stabilization task (crossover frequencies from 2 to 4 rad/sec) of attitude control for which the expression of Eq. 7 is adequate. Since the outer loop, usually referred to as the "path loop," is closed at a relatively low frequency where lead is difficult for the pilot to generate (on the order of 0.2 to 0.4 rad/sec), a pure gain pilot model is used. For the position closure, the general pilot transfer function is

$$Y_{p_x} = K_{p_x} e^{-\tau_e s} \tag{13}$$

In the closed-loop analyses to follow, control situations are reviewed, and the resulting implications on handling problems are correlated with experimental data, when available, to give metrics which will aid in the

unification of helicopter handling qualities. The multiloop situations of hover over a spot and lateral-directional approach are considered in the absence of external disturbances to yield information on damping requirements and good dynamic stability characteristics in these two critical flight regimes. These same two exercises are repeated in the presence of gusts to give further information on control activity, additional damping to cope with gusts, and the effects of other stability derivatives.

SECTION III

CLOSED-LOOP ANALYSES

The pilot-vehicle systems for three broad areas of closed-loop control are analyzed to determine the critical handling qualities factors in longitudinal and lateral-directional control situations. The piloting problems selected are mostly those situations where some experimental data are available in terms of the stability derivatives (dynamic characteristics) and pilot opinion or comments. The most representative of these problems are encountered in the performance of an ILS approach or in hovering over a spot. Various measures of the handling qualities difficulties were considered as appropriate, depending on whether system performance was good or bad, and these were correlated with pilot opinion. From these correlations the criteria for good handling qualities can be established. Although it is optimistic to generate generally applicable criteria from one or two sets of data, it appears, because of the strong theoretical backing, that major influences can be described in fairly simple terms.

The three situations of interest are:

- Hover over a spot
- Lateral-directional control during approach
- Longitudinal control during approach

The coverage given these three situations includes the effects of stability derivative variations on the open-loop factors (Section IV) as well as on the closed-loop system and on the pilot's closed-loop performance in the presence of gust inputs.

A. HOVER OVER A SPOT

The longitudinal dynamics of a hovering vehicle in pitch are completely specified by Eq. 3 and its five stability derivatives. In the handling qualities analysis of the closed-loop situation, the effects of control

sensitivity M_{δ_B} are not considered, and the resulting conclusions are only valid for situations in which M_{δ_B} is adjusted to its optimum value, i.e., degradations in pilot rating due to too high or too low a control sensitivity are not considered to begin with. Of the five derivatives, M_q and M_u are recognized as the most important. Consequently, the major emphasis will be on the effects of these two derivatives; that X_u and $X_{\delta_B}/M_{\delta_B}$ are of secondary importance is demonstrated in Appendix B. Figure B-1 of Appendix B illustrates the effects of M_u and M_q on the characteristic roots for a tenfold change in the two derivatives taken one at a time. Increasing M_u increases the phugoid frequency at nearly constant damping ratio and increases $1/T_{sp}$. Increasing the pitch damping (M_q more negative) increases the phugoid damping at roughly constant frequency and also increases $1/T_{sp}$. Additional information on the effects of the derivatives on the open-loop dynamics is readily obtained via the approximate factors of Appendix A.

1. Pitch Closure $\theta \rightarrow \delta_B$

In Section II-C the hover pitch closure for the simple case where $M_u = 0$ was given as an example. For the more general case ($M_u \neq 0$), where the control mode is characterized by the hover cubic, the vehicle transfer function $Y_c = \theta/\delta_B$ can be written from Eq. 3.

$$\frac{\theta}{\delta_B} = \frac{M_{\delta_B}\left(s + \frac{1}{T_{\theta_1}}\right)}{\left(s + \frac{1}{T_{sp}}\right)\left(s^2 + 2\zeta_p\omega_p s + \omega_p^2\right)} \tag{14}$$

where

$$\frac{1}{T_{\theta_1}} = -X_u + \frac{X_{\delta_B}}{M_{\delta_B}} M_u$$

and approximations for the cubic factors are given in Appendix A.

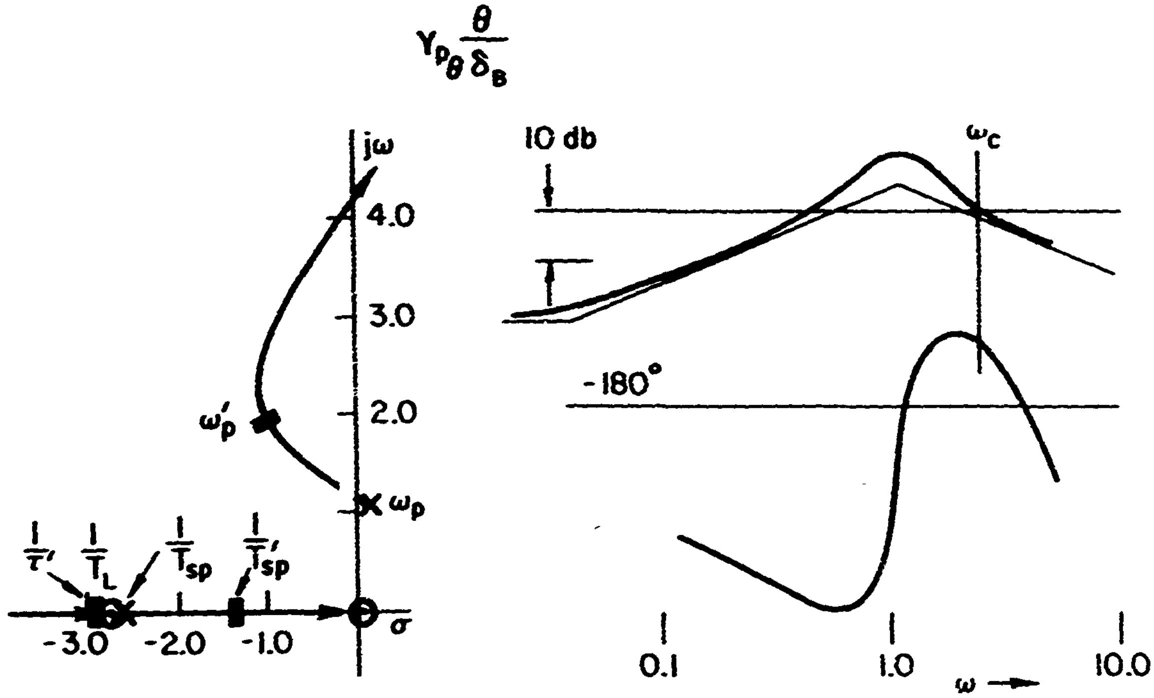
The pilot closure of $\theta \rightarrow \delta_B$ for the general cubic case of Eq. 14 is much the same as that for the simple hover expressions of Eq. 9.

To understand the similarity compare the $M_u = 0$ case of Fig. 2 with the $M_u \neq 0$ cases of Fig. 5. In the latter cases, increasing pilot gain K_{p_θ} increases the damping of the unstable phugoid ω_p for low values of gain, K_{p_θ} (the symbol ▮ locates the closed-loop poles). The effect of M_u is illustrated by comparing the pilot closure of $\theta \rightarrow \delta_B$ for the two configurations of Fig. 5 (cases 2 and 11 of Ref. 11). The (short-period) aperiodic modes, $1/T_{SP}$, associated with the damping, M_q, for both cases are in the good region with regard to pilot lead requirements ($1/T_{SP} > 1$, see example of Section II-C). Accordingly the pilot will select his crossover frequency from 2–3 rad/sec in a K_c/s region, with his lead frequency, $1/T_L$, approximately equal to $1/T_{SP}$. The Bode plots of Fig. 5 show that the frequency characteristics of the two situations are practically identical in the region of crossover. This suggests that the pitch closure is not critical with M_u in the <u>absence</u> of gusts. The experiments of Ref. 11 verify this result in that M_u is not important from a dynamic standpoint unless <u>trim effects</u> are included.

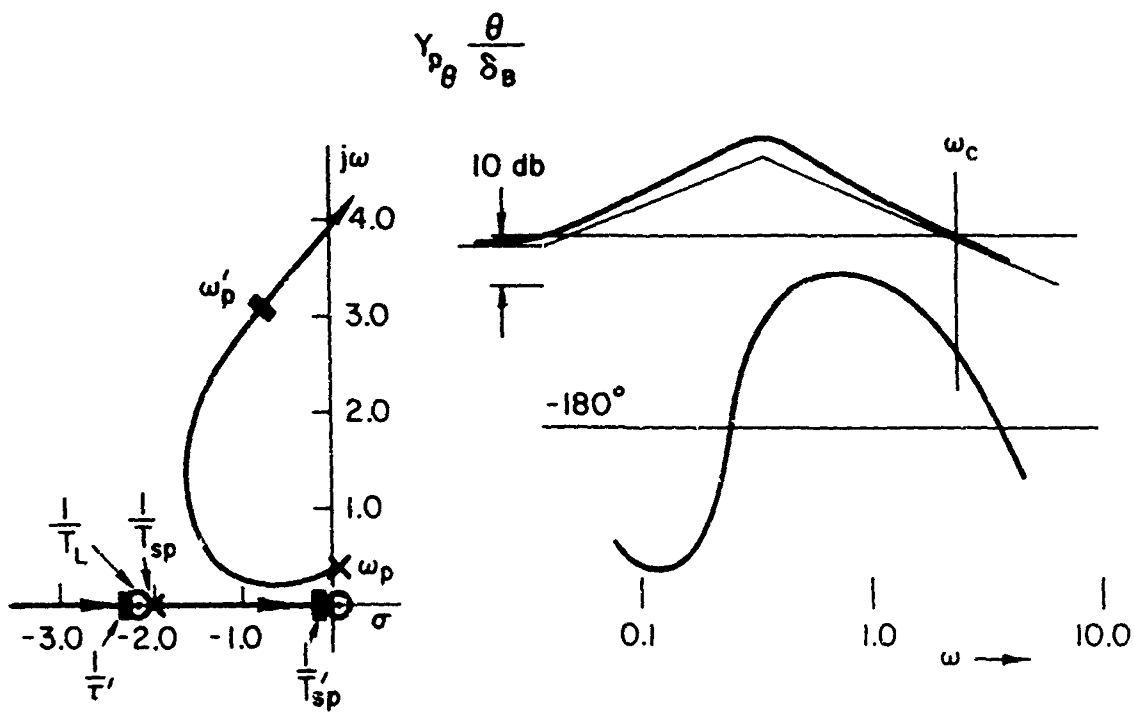
One significant effect of pitch damping, M_q, on the inner-loop closure is the same as given by the simple example case of Section II-C. If the damping, M_q, is increased above a minimum corresponding to $1/T_{SP} > 1.0\ \text{sec}^{-1}$, the pilot lead requirements are practically eliminated. The reduction in pilot lead requirements will be accompanied by an improvement in pilot's rating, provided system performance is good and the pilot gain required for crossover is "optimum."

2. Position Closure $x \rightarrow \delta_B$

The performance obtained in the position or "x" loop closure, Fig. 6, is a strong function of the kind of dynamic characteristics attainable with the closure of the inner, attitude loop. Most important of these is the achievement of positive damping of the phugoid and a large $1/T'_{sp}$ (the prime indicates the factor after the first closure). The latter refers to the location of $1/T'_{sp}$ with respect to the crossover frequency. Since the outer loop is closed at a relatively low frequency, it is difficult for the pilot to generate low frequency leads and such action would, in any event, produce a poor rating. It is easier for the pilot to adapt lead



a) High M_u



b) Low M_u

Figure 5. Pitch Hover Closures, $\theta \rightarrow \delta_B$

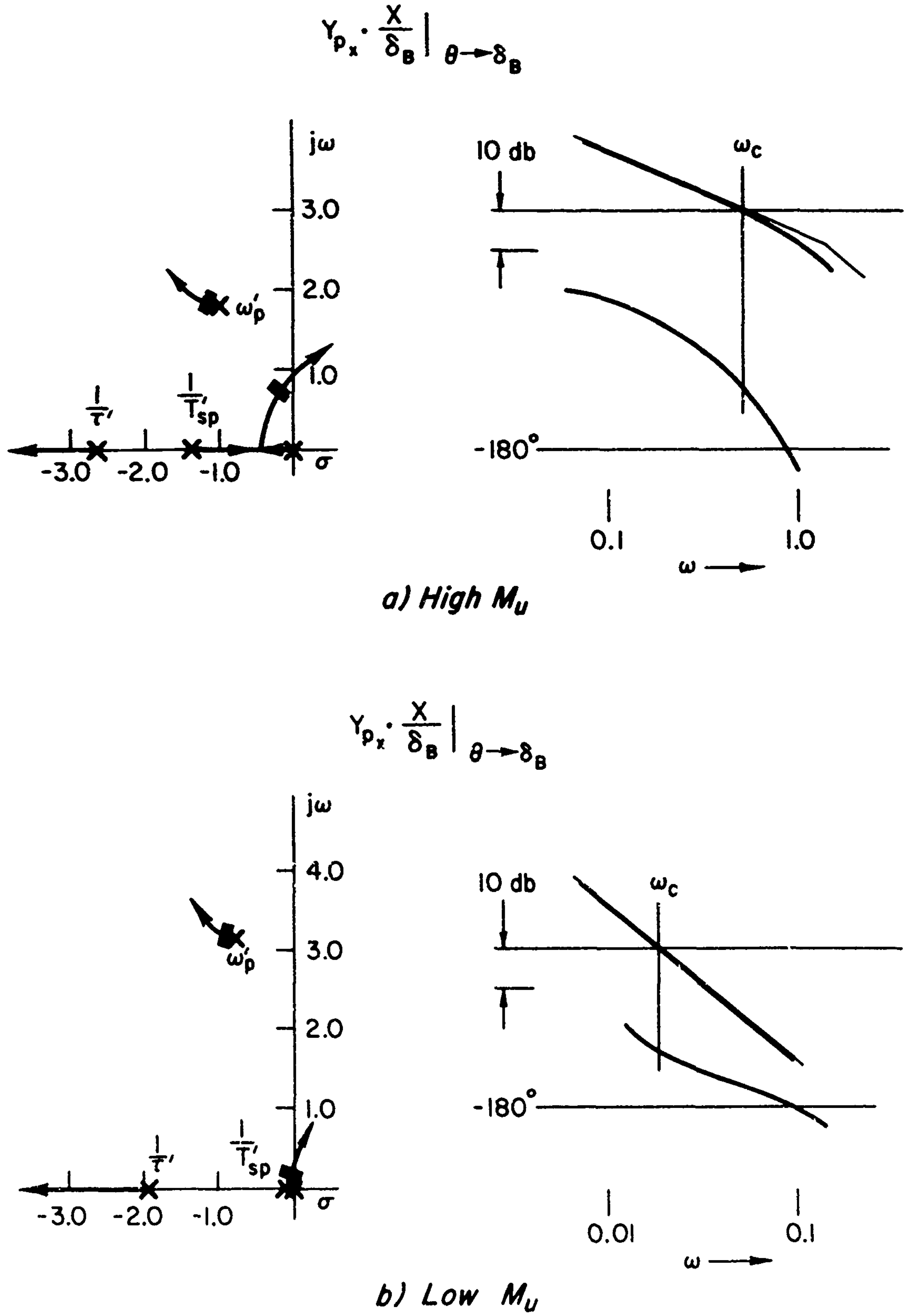
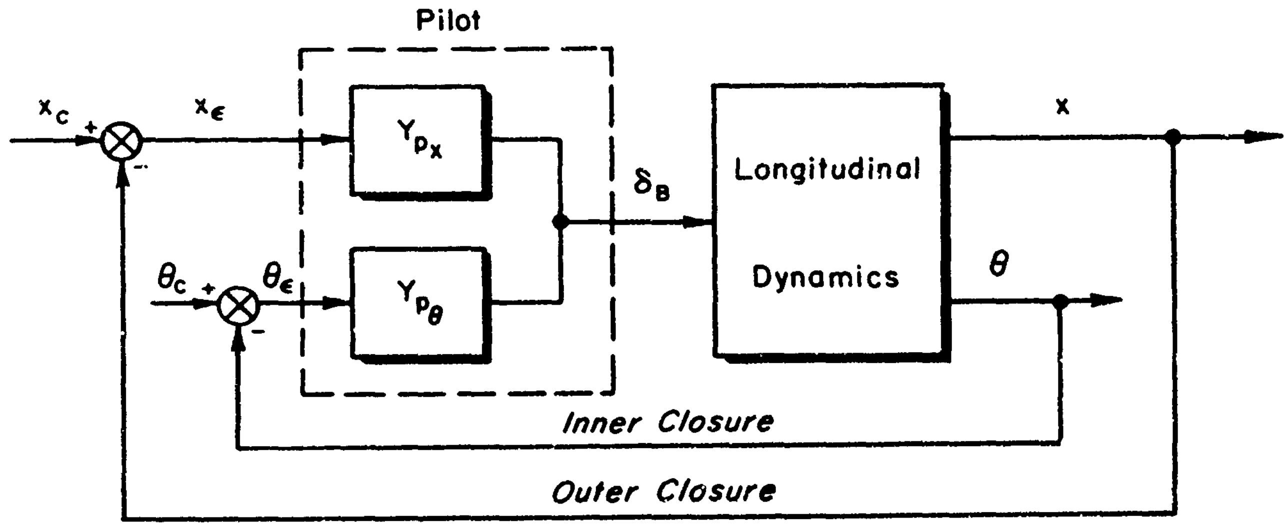


Figure [illegible]. Position Hover Closures, $x \to \delta_B$

in the faster inner loop than in the slower outer loop. On the basis of these observations, we can safely assume that lead is not used in the outer loop. The difficulties a pilot would have if he did not choose to close his attitude loop when attempting to hold his position over a spot is best illustrated by Fig. 7. In the sketch of Fig. 7a, it is impossible for the pilot to stabilize his phugoid mode without adapting more lead than he can easily accommodate. This was experimentally shown to be the case in Ref. 12 where the pilot was unable to "hover" the simulator given only position information. Having stabilized the phugoid with an attitude closure (Fig. 7b), the controlling modes of interest become the low frequency poles near the origin; and the ω_p modes associated with high inner-loop gain (Fig. 5) are little affected by the low gains used in the position closure.

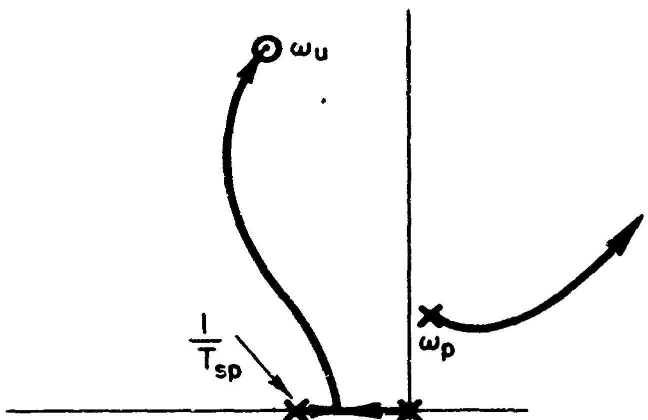
With regard to the specific effects of stability derivatives, the performance obtained in the displacement closure is probably more dependent on M_u than on the other derivatives. As shown in Fig. 5, the large $1/T'_{sp}$ is associated with the high M_u. Figure 6 gives the outer-loop closure for the two inner-loop cases of Fig. 5. The low M_u case of Fig. 6b is an example of a K/s^2 situation where the small $1/T'_{sp}$ combines with the free s of the "x" closures. The pilot would be critical of this situation if higher performance tracking were required, since the achievable bandwidth would be too low to cope with fast inputs without the pilot supplying large amounts of lead. The tracking performance of the high M_u case, Fig. 6a, is improved due to the higher system bandwidth and the K/s characteristics in the region of crossover. The pilot can now achieve a desired crossover frequency without going unstable and at the same time obtain lower rms position errors (see Fig. 9). A minimum desirable outer-loop crossover frequency of approximately 0.3 rad/sec is indicated in the Appendix B analyses. At this time there is very little experimental data to support this value beyond the limited data of Ref. 17.

In a similar way, the larger the closed-loop factor $1/\tau'$ (Fig. 6), the smaller its phase margin contribution in the position closure. This factor is set by $1/T_{sp}$ (Fig. 5) and therefore responds favorably to either increased M_u or M_q, as shown in Fig. B-1. Raising the minimum $1/T_{sp}$ above about 1.5 (based on 0.3 rad/sec crossover) will probably benefit the performance of

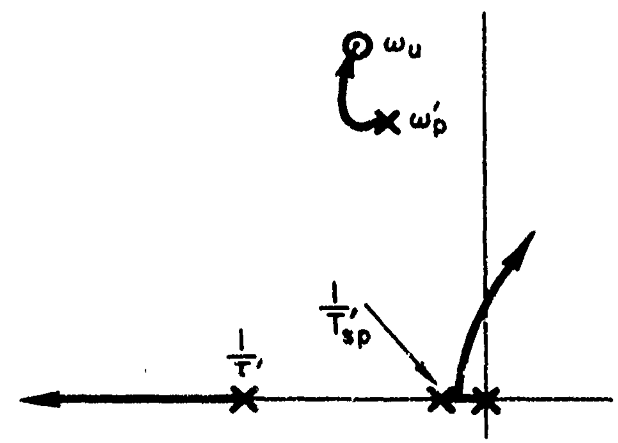


$$\left.\frac{x}{\delta_B}\right]_{Y_{p_\theta}=0} = A_u \frac{\left(s^2 + 2\zeta_u\omega_u + \omega_u^2\right)}{s\left(s + \frac{1}{T_{sp}}\right)\left(s^2 + 2\zeta_p\omega_p + \omega_p^2\right)}$$

$$\left.\frac{x}{\delta_B}\right]_{\theta\rightarrow\delta_B} = A_u \frac{\left(s^2 + 2\zeta_u\omega_u + \omega_u^2\right)}{s\left(s + \frac{1}{T'}\right)\left(s + \frac{1}{T'_{sp}}\right)\left(s^2 + 2\zeta'_p\omega'_p + \omega_p'^2\right)}$$



a) No Inner Loop Closure



b) After Inner Loop ($\theta \rightarrow \delta_B$) Closure

Figure 7. Typical Longitudinal Hover Closure

outer closure, but this requires experimental verification. However, increasing separation of both the $1/\tau'$ and $1/T'_{sp}$ modes from the low frequency control mode of the position closure will always improve the phase margin.

Another factor which can make a significant contribution to the outer-loop performance is the zero, $1/T_{\theta_1}$, of the pitch closure. If the zero had a minimum value of 0.3 to 0.5 rad/sec, it would guarantee a K/s characteristic in the desired crossover region of the $x \rightarrow \delta_B$ closure. This can be seen in Fig. 8 where the zero, $1/T_{\theta_1}$, represents the minimum value that $1/T'_{sp}$ can have; in turn, this $1/T'_{sp}$ sets the maximum outer loop crossover frequency for a pure gain closure — provided all other factors are optimally located. The usual approximate factor for this zero is given in Eq. 16 where, for the single rotor, X_δ is nonzero but $1/T_{\theta_1}$ is still very small (near zero). In the case of a tandem employing only differential collective pitch for the longitudinal stick control, the factor becomes equal to $-X_u$. However, large X_u's derived aerodynamically will not be as satisfactory as those obtained through inertial feedbacks (see Ref. 12) because of the detrimental gust effects. Since special means (inertial sensors) are required to obtain a good location of the $1/T_{\theta_1}$ factor, this criteria would not be practical in specifying good handling qualities.

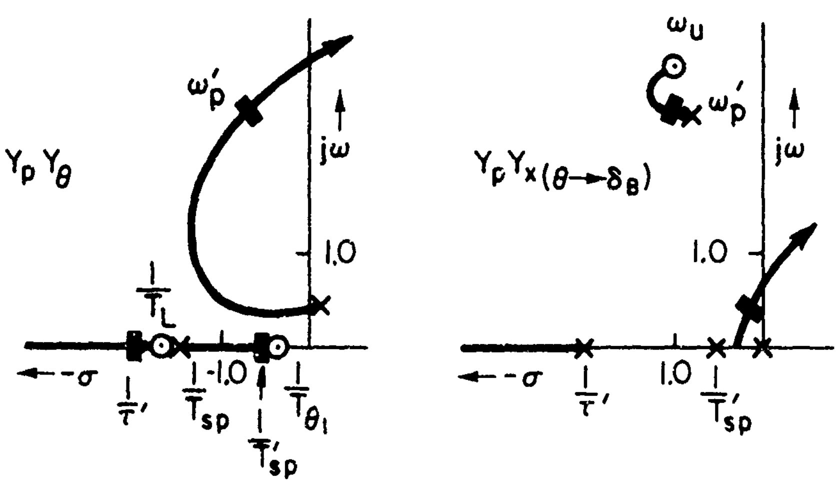


Figure 8. Effect of the Zero, $1/T_{\theta_1}$, on the Outer-Loop Closure

3. Hover Steadiness Without Gust

The ability of the pilot-helicopter combination to hover over a spot within a given maximum control deflection is currently specified as a measure of the handling qualities at hover. This criterion is difficult to analyze since the inputs to the pilot-vehicle system are not easy to identify or describe. In the absence of external gusts, the inputs must be considered to be either internal to the pilot and the helicopter, such as the pilot's own noise or remnant, or due to aerodynamic turbulence generated in the rotor downwash.

The use of the analog fixed-base simulator is about the best means of obtaining some representative data in this area. The results of Fig. 9 show the pitch, displacement, and control responses to the internal noise input of a human pilot in a fixed-base simulated hover. The figure demonstrates the M_u effect on the outer-loop closure. The "x" error was reduced by a factor of three for a tenfold increase in M_u. The improved "x" performance for the high M_u case was predicted qualitatively in the study of the outer-loop closure. It is also important to note that the control responses were also reduced by about the same factor as the "x" displacement error. This presents a potential danger if the specification is interpreted to mean that the smaller the control deflection in still air, the better the helicopter handling qualities. As proved in Ref. 11, the pilot would rate the two configurations of Fig. 9 the same. This means that the pilot in these tests is willing to trade performance (response errors) for stability. The high M_u case has better closed-loop hover performance, as already shown, but the low M_u case has less negative damping of the phugoid mode and is therefore more stable open-loop. Also, in the presence of gusts the low M_u case is superior because of its lower gust sensitivity. This is the primary subject of the following section, but the point to be made here is that the pilot may be more critical of the handling qualities in the presence of gusts than to his hover performance in still air.

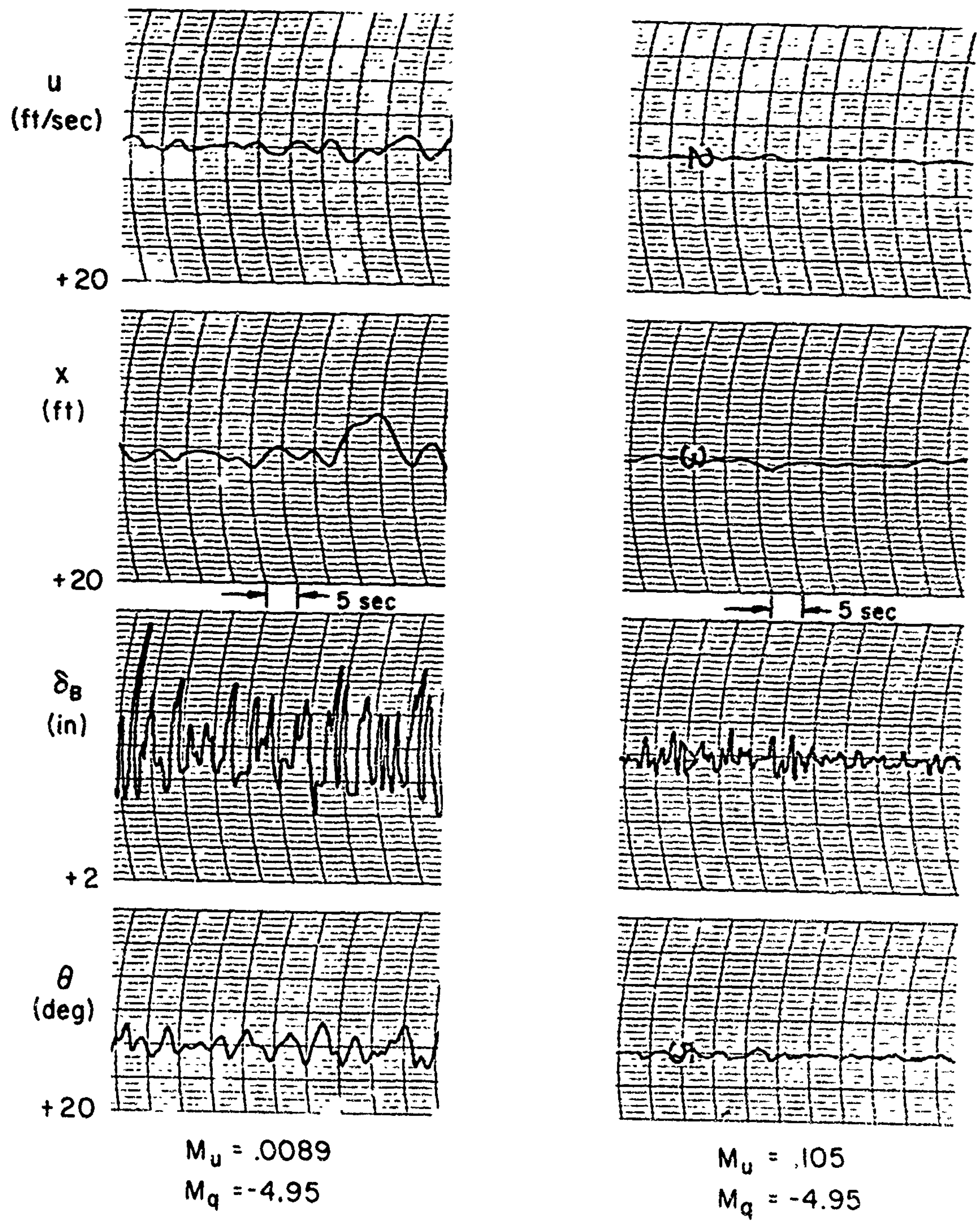


Figure 9. The Effect of M_u on Hover Steadiness in Still Air

4. Effects of Gust Disturbances on Closed-Loop Hover Task

The preceding handling qualities discussions have to do with a pilot hovering in <u>still air</u>. When the above task includes gusty air, then the pilot is also concerned with the variations in position, attitude, and control deflection as influenced by the gust disturbances. The purpose of this section is to discuss the results of a rather complete analysis of hover in the presence of random u-gust disturbances. The detailed study which examines the effects of gusts on pilot closures is presented in Appendix B. The results presented here are primarily concerned with the effects of changes in pilot pitch and position closure gains and of changes in the stability derivatives on the rms x, θ, and $M_{\delta_B}\delta_B$ responses with <u>both</u> loops closed.

The main point of the hover analysis contained in Appendix B is an examination of four combinations of derivatives; two values of both M_u and M_q. The two values for each derivative differ by a factor of ten and the total of four values selected adequately covers the range of the critical root positions (see Fig. B-1 of Appendix B).

For the hover work the gust spectrum is given by Eq. B-1, where the gust break frequency, $\omega_g = (3/2)(V_{as}/L)$, was chosen as 1.0 rad/sec (this corresponds to an L of 30 ft and a mean wind speed of 20 ft/sec). The rms gust responses were not found to be strongly dependent on ω_g.

a. **<u>Effect of Gains</u>**. The rms values of σ_x, σ_θ, and $M_{\delta_B}\sigma_{\delta_B}$ were determined for variations in the θ and x loop gains with both loops closed, but only one gain varied at a time, with the other held fixed at the nominal values. The sketch of Fig. 10 shows typical variations in the σ's as a function θ and x closure gains about the nominal. The nominal gains were selected according to the pilot adjustment rules for good handling qualities as outlined in Section II. It can be seen in Fig. 10 that these nominal gains provide a reasonable compromise as to a composite minimum among the various "σ" values. The most significant result shows σ_θ and σ_x a decreasing function of their respective gains, K_θ and K_x; however, the remaining σ's in each case are usually an increasing function of the loop gain, e.g , the pilot's control utilization, $M_{\delta_B}\sigma_{\delta_B}$, and the position response, σ_x, increase with "θ" gain as do the pitch response, σ_θ, and $M_{\delta_B}\sigma_{\delta_B}$ with "x" gain.

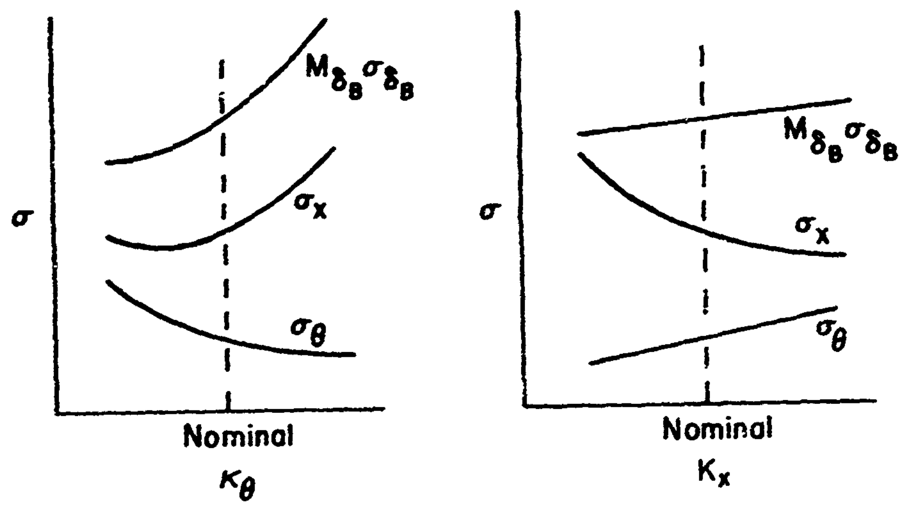


Figure 10. Sketch of σ Variations About Nominal Gains

b. __Effect of the Stability Derivative__. Having verified the suitability of the nominal pilot gains, the effects of changes in the two most important stability derivatives M_u and M_q on the rms responses can be analyzed. Figure 11 displays the analytical performance results obtained as well as the experimental pilot rating data of Ref. 11. The four analytical cases are defined as follows:

	VALUE OF DERIVATIVE	
	M_u(1/ft-sec)	$-M_q$(1/sec)
Low M_u, Low M_q	.0088	0.15
High M_u, Low M_q	.088	0.15
Low M_u, High M_q	.0088	1.5
High M_u, High M_q	.088	1.5

The analytical results (Fig. 11a) show little variation of the σ's with respect to M_q except for $M_{\delta_B}\sigma_{\delta_B}$ at the high M_u. This result is consistent with the flight test data of Fig. 11b, i.e., pilot opinion rating (PR) __is not__ a strong function of M_q. This follows also from the Fig. 3 trends and the observation that the range of tested M_q's in Ref. 11 corresponds to values of $T_{sp} < 1$, where the pilot lead adaptation is minor. For the smaller values of M_q analyzed, the results of Fig. 3 would indicate a sharp increase in rating; but the analysis (Fig. 11a) shows no general decrement in performance. This result is in line with the Section II discussion where we concluded that the rating increment should be ascribed to changes in pilot equalization, T_L, rather than to system performance, and that significant changes in T_L would occur only for $T_{sp} > 1$.

Note: Top graphs only show trends–variations between data points are unknown

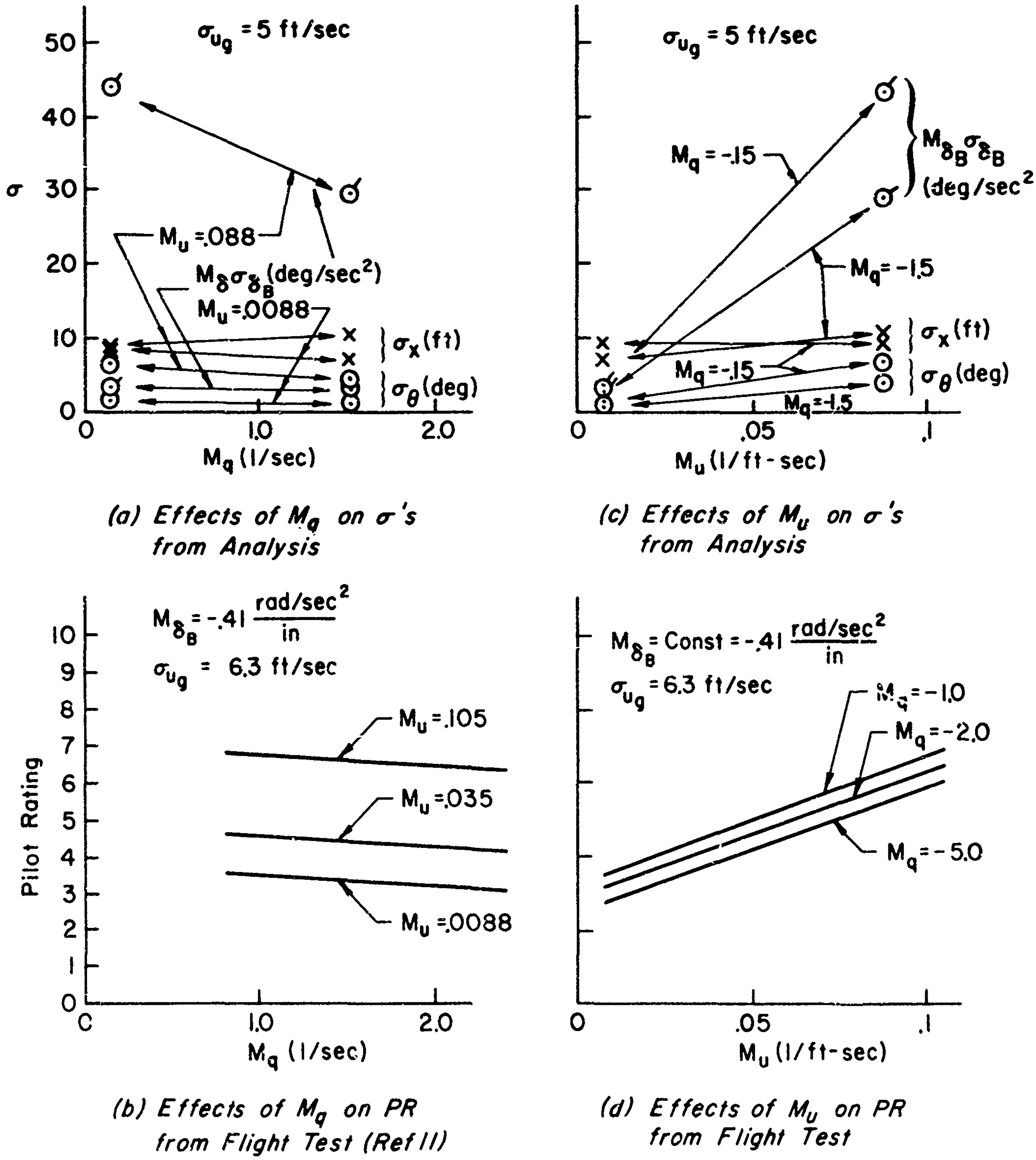


Figure 11. Effects of Stability Derivatives M_u and M_q on RMS Responses; and Correlations with Experimental Data

The effect of changing M_u by a factor of 10 is shown by plot "c" of Fig. 11, and the results can be summarized as a negligible effect on σ_x, a large change in σ_θ (roughly a factor of 3), and a very large change in $M_{\delta_B}\sigma_{\delta_B}$ (roughly a factor of 10). The latter change appears to be directly proportional to M_u. In fact, an approximate formula relating the response spectra in $M_{\delta_B}\delta_B$ to the gust input spectrum (see Appendix B-I) is simply

$$M_\delta \sigma_\delta \doteq 1.4 M_u \sigma_{u_g} \text{ rad/sec}^2 \tag{15}$$

Accordingly, the control power required to combat the maximum gust expected (4σ value) is easily estimated as

$$(M_\delta \delta)_{max} \doteq 4 M_\delta \sigma_\delta \doteq 5.6 M_u \sigma_{u_g} \tag{16}$$

Thus, for the values of M_q tested, M_u emerges as the more significant parameter on the basis of both the analytical results and the experimental evidence in Fig. 11d.

Combining the results of plots "c" and "d" of Fig. 11 suggests a strong correlation between the flight test pilot's opinion rating data and the computed pilot's control activity, $M_{\delta_B}\sigma_{\delta_B}$, in the presence of gusts. This is consistent with the functional relationship of Eq. 8 and the observation that, for the test conditions, neither performance, σ_x and σ_θ, nor equalization, T_L and T_I, variations occur; therefore, pilot rating should be reflected in control activity, σ_δ. This is also suggested by Ref. 13, where the measured control response, σ_δ, is roughly related to the pilot rating for constant M_δ. But if σ_δ does reflect pilot rating, then a desirable or optimum level of M_δ (corresponding to a given optimum value of σ_δ, say) should be linear with M_u according to Eq. 15. This expected result is confirmed by the experiments of Ref. 14, where the optimum sensitivity is defined on an $M_q - M_\delta$ plot for a number of gust sensitivities, M_u, and for a representative gust. The limited results of the experiments in Ref. 14 show that the optimum control sensitivity, $M_{\delta_{opt}}$, is a function of the damping, M_q, the velocity stability, M_u, and the value of X_u which has a direct effect on $1/T_{\theta_1}$ (Eq. 14). For values of $-X_u$ appropriate to helicopters (generally less than 0.05) and for the ranges of M_q and M_u covered, the optimum value of M_δ is fitted by the approximate relationship

$$M_{\delta_{opt}} \doteq 0.23 - 0.03 M_q + 6 M_u \ \frac{\text{rad/sec}^2}{\text{in.}} \tag{17}$$

for $\quad 1 < -M_q < 6 \quad ; \quad 0 < M_u < 0.031$

The expected variation with M_u is clearly shown by this relationship. The variation with M_q is "explained" later in connection with the open-loop analyses of Section IV.

5. Summary of Important Analytical Results for Hover Over a Spot

a. To reduce the pilot's lead requirements to where they have a negligible effect on rating, the minimum damping about all axes in the absence of gust should be greater than 1 sec^{-1} (p. 13). For multiloop control such as attitude stabilization and hovering over a spot, the minimum damping should probably be increased to 1.5 sec^{-1} (p. 21).

b. For pitch control in still air, M_u is relatively unimportant from a dynamic standpoint; increasing M_u improves the ability of the helicopter to hover over a spot (still air only) by providing a larger crossover frequency in the position closure (pp. 18, 21, 24).

c. Increasing the numerator factor $1/T_{\theta_1}$ through nonaerodynamic augmentation improves the precision hover performance, σ_x, in both still and gusty air (p. 22).

d. Selecting θ and x pilot-closure gains according to established "adjustment" rules provides a good composite minimum to σ_θ, σ_x, and σ_{δ_B} responses in gusty air (p. 26).

e. In gusty air the rms control response, σ_δ, is roughly proportional to $M_u \sigma_{u_g}$ (p. 29).

f. An increase in M_q above the minimum of Item a produces small favorable changes in attitude responses, σ_θ, and control deflection, σ_δ, in gusty air (p. 29).

g. Correlation of pilot rating data in hovering over a spot with analytical results of the attitude and position manual closures indicates that pilot rating is primarily a function of control activity, σ_δ, provided the Item a condition is met (p. 29).

h. Analysis of recent experiments shows the optimum control sensitivity, M_δ, to be a function of M_u, as well as M_q, in the presence of gusts. The optimum M_δ is a simple function of both damping, M_q, and the velocity stability, M_u (p. 29).

i. The control power required to cope with the maximum gust expected can be specified on the basis of the rms control deflections for random gust inputs, for example (p. 29),

$$\delta_{max} = 4\sigma_\delta$$

j. All the foregoing apply also to lateral and directional control in hover with appropriate modifications in the motion quantities and derivatives, i.e.,

$$M_q \rightarrow L_p \rightarrow N_r \qquad M_\delta \rightarrow L_\delta \rightarrow N_\delta \qquad x \rightarrow y$$

$$M_u \rightarrow -L_v \rightarrow N_v \qquad \theta \rightarrow \varphi \rightarrow \psi$$

B. LATERAL-DIRECTIONAL CONTROL DURING APPROACH

The in-flight model simulator experiments of Refs. 15 and 16 study the lateral-directional handling qualities of tandem-rotor and single-rotor configurations, respectively. In some cases the range of parameters is great enough that the characteristics of the simulators are overlapping. The two experiments were designed to determine the requirements on directional speed stability, N_v, and damping, N_r, but the flight environment and the piloting techniques were different enough to require two separate analyses. In the NASA test (Ref. 15) the pilot task was to null the localizer and glide slope indicators on an ILS approach, while in the NRCC tests (Ref. 16) the pilot maintained a ground track and glide slope through visual alignment of a ground-fixed indicator. In the face of the varied conditions of the simulators and test conditions, our approach has been to analyze appropriate closed loops with and without external disturbances and to correlate the data with the appropriate analyses.

In the first section the lateral-directional analysis without gust disturbances will be performed and correlated with those experimental data which are not strongly influenced by gusts. Under this category the dynamics of the airframe are the primary influence on the pilot's rating on the handling qualities. In order to nullify the influence of the large canned gust inputs (through N_v) to the NRCC experiments, only those configurations with the lowest value of N_v are considered to be correlative with the NASA tests. Because of the large differences in geometry and weight, it is instructive to find dynamically similar situations (i.e., same transfer function factors) to compare NASA and NRCC pilot ratings.

1. Control of Heading With "Ailerons"

To track the localizer beam through deviations displayed on his ILS instrument, the pilot closes the loop by maneuvering the airplane to zero the deviation, preferably by using "ailerons" alone to make corrections.

The control of lateral offset and heading through aileron is considered long term or outer loop control; and since it is generally impossible for the pilot to follow the proper ground track by just "zeroing" the meter deviation, he must provide a good speed of response in turning through

another aileron closure of the roll inner loop. The lateral-directional loop ($y \rightarrow \delta_A$) is not considered as a separate factor here because this closure is normally of very low bandwidth which is set primarily by the bandwidth of the heading closure (Ref. 17). If the aircraft has good heading control, the lateral deviation control will be good. Accordingly, two basic loop closures are considered in evaluating the handling qualities of the various configurations:

a. Bank angle to lateral cyclic stick, $\varphi \rightarrow \delta_A$
b. Heading to lateral cyclic stick, $\psi \rightarrow \delta_A$

The criteria used for closing the $\varphi \rightarrow \delta_A$ loop are simple and involve the pilot's desire to obtain good system performance with minimum lead, and at the same time achieve good damping of the dutch roll as shown in Fig. 12a. As for the ideal pitch case of Section II-C, the $\varphi \rightarrow \delta_A$ closure will involve pilot lead, T_L, adaption which effectively cancels the roll subsidence mode, i.e., $T_L \doteq T_R$ (see Fig. 12a). In these experiments the roll damping was not on trial ($-L_p \doteq 1/T_R$) because T_R was always on middle ground, $0.25 < T_R < 0.67$; therefore, selecting $1/T_L$ identically equal to $1/T_R$ was not critical and provided a good basis for comparison. In fact, there is no improvement in rating for T_R below 0.5 to 1.0 as borne out by the discussion of Section II-C.

The important effect to be demonstrated here is the influence of the numerator damping, $\zeta_\varphi \omega_\varphi$, which is just beginning to be recognized (Ref. 3) as a basic metric of heading control performance not necessarily limited to helicopters. The oscillatory numerator case of Fig. 12a is typical of many of the NASA configurations tested in Ref. 15 where $\zeta_\varphi \omega_\varphi$ was always greater than or approximately equal ($L_v = 0$) to $\zeta_d \omega_d$. The $\zeta_\varphi \omega_\varphi$ effect can be appreciated by studying the heading closure $\psi' \rightarrow \delta_A$, where the prime again indicates that the inner (φ) loop has been closed. Referring to the heading closure of Fig. 12a, it can be seen that the maximum damping that ω_d' can achieve depends on $\zeta_\varphi \omega_\varphi$ of the roll numerator; and for the roll-closure gain required (indicated by the symbol ▮) to give a crossover of approximately 2 to 3 rad/sec in the inner loop, $(\zeta_d \omega_d)'$ of the heading closure approximately equals $\zeta_\varphi \omega_\varphi$. The maximum usable gain in the heading closure is limited in this case by stability considerations of ω_d'' (i.e., having adequate gain and

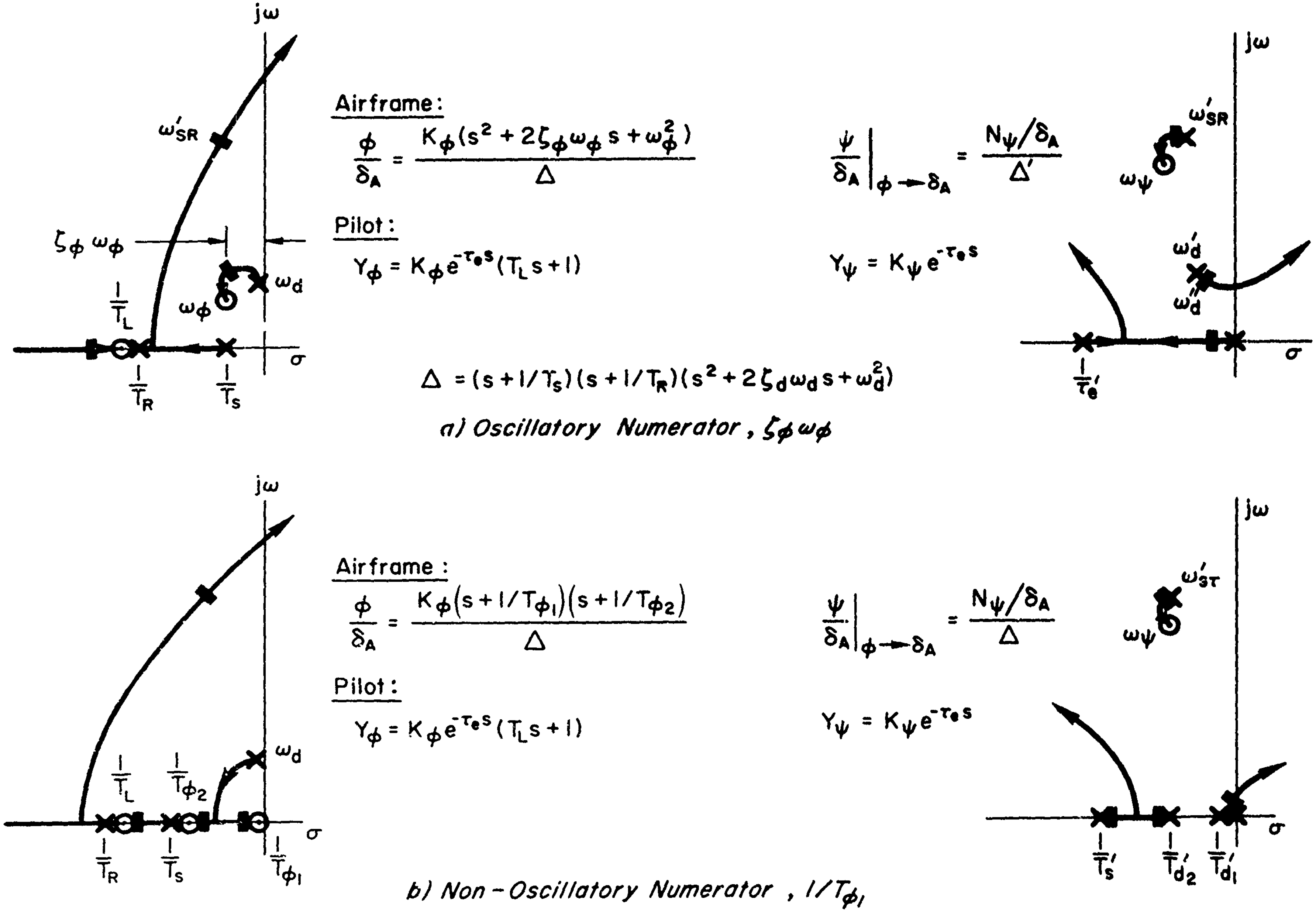


Figure 12. Typical Multiple-Loop Closures for Roll and Heading Control by δ_A

phase margins). More specifically, the extent to which the heading gain (and crossover frequency) can be increased depends on the value of $\zeta_d'\omega_d'$, which in turn is most strongly influenced by the basic value of $\zeta_\varphi\omega_\varphi$. In effect, the heading loop crossover frequency (or bandwidth or dominant response frequency) is determined by the value of $\zeta_\varphi\omega_\varphi$, which in turn is approximately given by (Appendix A)

$$\zeta_\varphi\omega_\varphi \doteq \frac{-Y_v - N_r}{2} \tag{18}$$

The other terms associated with this approximation (such as N_p, L_r, L_{δ_r}, and N_{δ_A}) are either negligible or were not present in the simulator.

Figure 12b shows a similar dependence of the achievable heading response speed (or crossover) on $1/T_{\varphi_1}$ for situations where the φ/δ_A numerator is nonoscillatory, i.e., approximately when $4N_\beta < (Y_v - N_r)^2$, most likely to occur for tandems, which have low directional stability, N_β (see Fig. 1). In this case the most critical loop in the heading closure is the closed-loop pole of the roll closure, which becomes almost equal to $1/T_{\varphi_1}$. As seen by the heading closure of Fig. 12b, this pole, $1/T_{d_1}'$, sets the crossover of the heading control. The approximate factor for this zero, for the above condition on N_β is

$$\frac{1}{T_{\varphi_1}} \doteq -Y_v + \frac{N_\beta}{Y_v - N_r} \tag{19}$$

Figure 13 shows the correlation obtained using the above basic φ-numerator characteristics as metrics. In this figure all the NASA data of Ref. 15 and three NRCC cases (lowest value of N_v = 0.01/ft-sec was used to nullify gust effect) of Ref. 16 are plotted to establish a boundary value of $\zeta_\varphi\omega_\varphi$ (or $1/T_{\varphi_1}$) $\doteq$ 0.4/sec associated with the 3-1/2 rating. This value for $\zeta_\varphi\omega_\varphi$ (or $1/T_{\varphi_1}$), which, as explained above, is to be taken as a measure of the heading response, converts to a crossover frequency between about 0.25 to 0.3 rad/sec. For crossovers less than 0.25 to 0.3 rad/sec, the pilot will begin to complain that the aircraft will not follow into a turn using "aileron" control. These results are in good agreement with the conclusions reached in Ref. 17 where the supersonic transport configurations studied are very much different in size and approach speed. In fact, the two in-flight simulators of Refs. 15 and 16 are themselves much different in size and weight

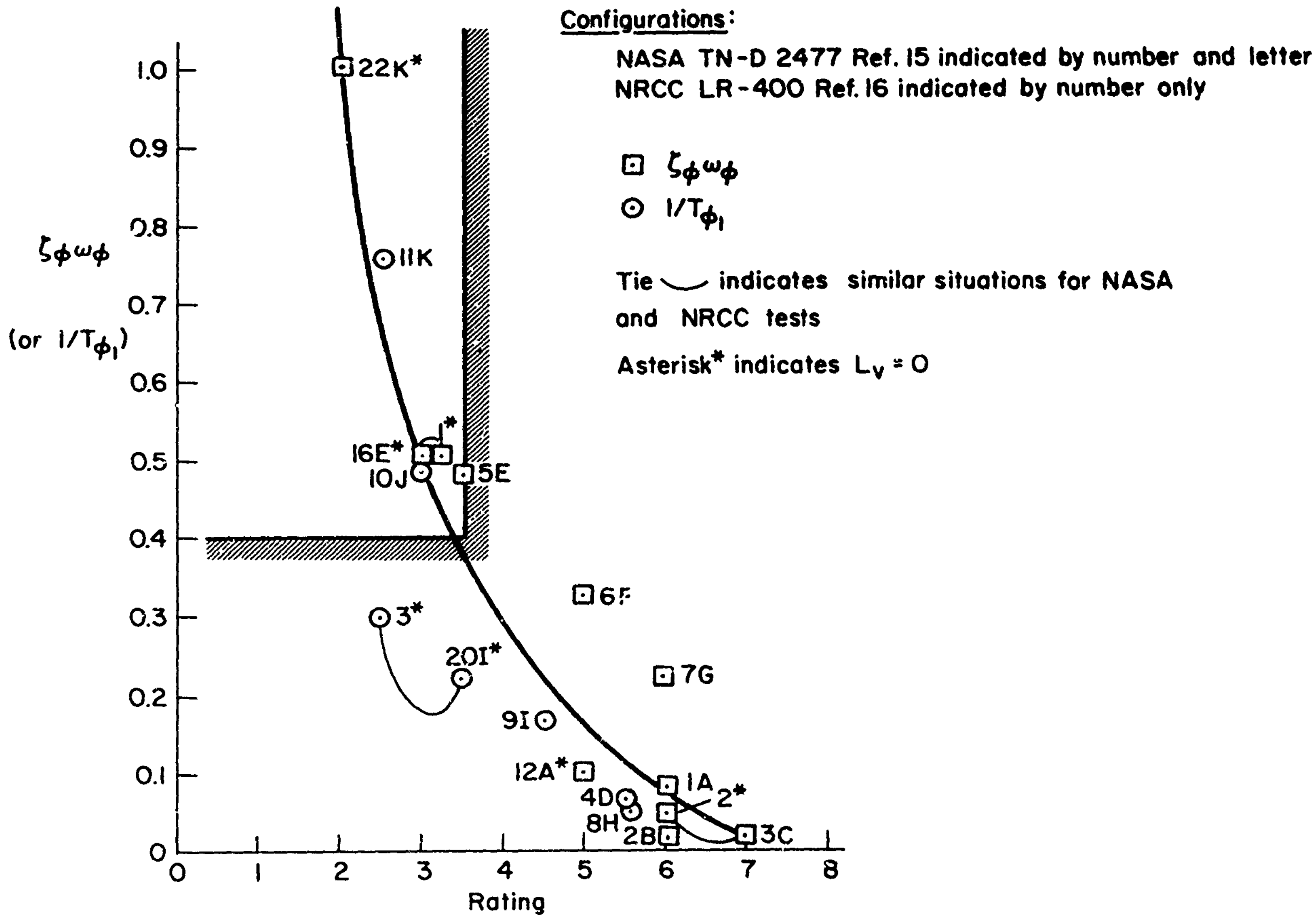


Figure 1?. Correlation of Lateral Numerator Damping Factor, $\zeta_\varphi \omega_\varphi$ (or $1/T_{\varphi_1}$), with Pilot Rating

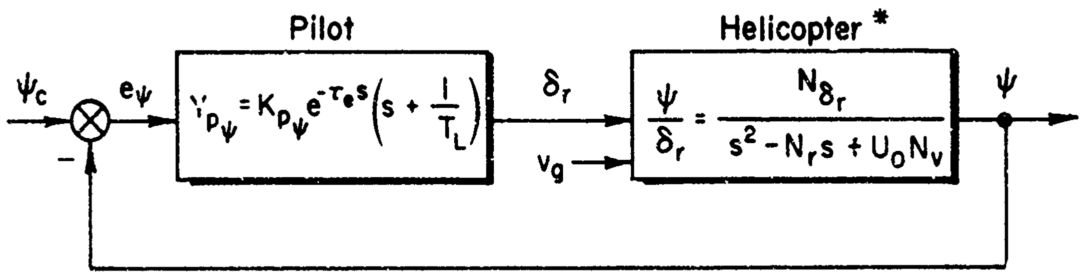
(see Appendix B, p. 147), but the pilot ratings of similar dynamic situations in the two aircraft gave surprisingly good agreement, as indicated in Fig. 13.

For a more detailed study of the foregoing results, Appendix B contains a configuration review of several representative situations taken from Refs. 15 and 16. Appendix B gives the closure details using the pilot models of Section II together with the transfer function data for the example cases. The pilot comments are also discussed with respect to the closure characteristics when these comments are available.

2. Control of Directional Disturbances With "Rudder"

In the experiments discussed above, the pilot was controlling his long term lateral deviation from a ground track which reduced to the control of heading with δ_A. Also, the influence of gust disturbances was slight. In this section the investigation is concerned with pilot control of gust-induced heading disturbances with "rudder," δ_r. Because the gust disturbances are only directional in nature ($L_v = 0$) and the vehicle's roll characteristics are suppressed (high $-L_p$), the pilot is interested primarily in the single-loop regulation of heading with pedals. As opposed to the previous study, the value of N_v is never less than 0.01, and increasing values of N_v change the disturbance level directly. The purpose here is to analytically determine the closed-loop performance and other factors for the approach situations of the NRCC data, and then to correlate the results with the pilot ratings.

The pilot-vehicle control loop has the form shown in Fig. 14.



$$* \begin{cases} 2\zeta_d\omega_d \quad \text{or} \quad 1/T_{d_1} + 1/T_{d_2} \doteq -N_r \ (Y_v \text{ very small relative to } N_r) \\ \omega_d^2 \quad \text{or} \quad 1/T_{d_1}T_{d_2} \doteq U_0 N_v \end{cases}$$

Figure 14. Heading Closure $\psi \rightarrow \delta_r$

The gust input spectrum of Eq. B-1 was a simple approximation to the actual "canned" gust used in the experiments when ω_g was chosen at 0.5 rad/sec.

Appendix B contains a detailed examination of twelve $\psi \rightarrow \delta_r$ pilot closures representative of the NRCC-approach test of Ref. 16. The twelve combinations contain five variations in yaw damping, N_r, and four variations in yaw speed stability, N_v.

a. **Effect of Gain**. Variations in pilot gain had much the same effect on the rms responses in ψ and $N_{\delta_r}\delta_r$ as was shown in hovering over a spot. A typical result is depicted in Fig. 15a where the σ_ψ response varies as an inverse function of closed-loop gain and the control response $N_{\delta_r}\sigma_{\delta_r}$ varies more or less as a direct function. As in the hover case, the nominal gain selected on the basis of the conventional "rules" produced a good compromise between performance, σ_ψ, and the pilot's control activity, σ_{δ_r}.

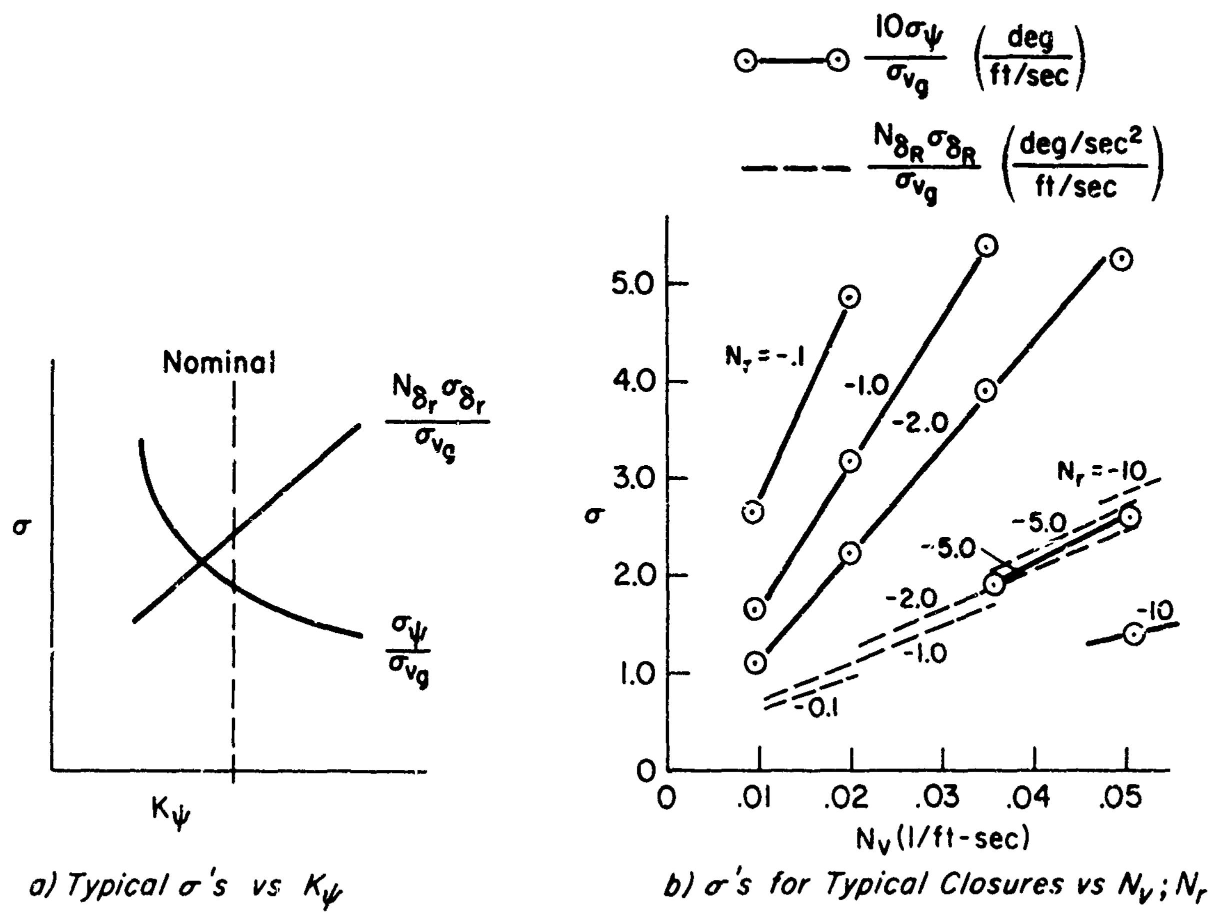


Figure 15. Analytically Derived Closed-Loop Performance

Figure 15b gives the analytically derived rms response in ψ and $N_{\delta_r}\delta_r$ as a function of N_v and N_r, the only two variables required to define the characteristic roots. These results show both σ's to be a strong function of N_v, but only σ_ψ to be a strong function and $N_{\delta_r}\sigma_{\delta_r}$ a weak function of N_r.

b. **Correlation with Experimental Data**. The analytical results are correlated with the experimental data in Fig. 16 to provide directional handling qualities criteria. The pilot rating cases shown in Fig. 16a are those for an "optimum" gain, N_{δ_r} (Table B-XI), which is the "best opinion" gain for the particular values of N_v and N_r. The computed values of σ_{δ_r} for these "optimum" N_{δ_r}'s and the computed values of σ_ψ (which are independent of N_{δ_r}) are plotted versus the corresponding experimental pilot ratings in Fig. 16b. On the basis of these relationships and the observation that lead adaptation is generally small (Table B-XI), it appears that the pilot may be equally sensitive to system errors, σ_ψ, and control activity, σ_{δ_r}. Since σ_{δ_r} is little affected by the value of N_r (Fig. 15b), it appears that the optimum value of N_{δ_r} is selected to reduce σ_{δ_r} to an acceptable level; i.e., for ratings better (less) than 3.5, σ_{δ_r} must be less than about 0.22 in. (Fig. 16b). The plot of Fig. 17 shows that such values of N_{δ_r} are essentially linear with N_v. However, the combinations of N_v and N_{δ_r} which produce the desirable level of σ_{δ_r} will not be considered satisfactory unless σ_ψ is also reduced to values below about 2° (Fig. 16b). But the only real influence on σ for a given N_v is N_r (Fig. 15b), which must therefore be inc. ased, with increasing N_v, to reduce σ_ψ to acceptable levels. The appropriate level of N_r must also be such to produce acceptable open-loop responses to control inputs (discussed later).

The similarities and differences between the above results and those obtained in the analysis of the longitudinal hover task are worth delineating. Both sets of calculations show that the rms attitude changes (σ_θ, σ_ψ) are similar functions of the damping (M_q, N_r) and speed stability (M_u) or directional stability (N_v); also that control activity ($M_\delta\sigma_\delta$, $N_{\delta_r}\sigma_{\delta_r}$) is primarily influenced by the gust excitation, which is proportional, respectively, to the speed and directional stability. However, the pilot's chief concern regarding system performance in the longitudinal case is σ_x, and this is not strongly influenced by either damping or speed stability. Therefore the σ_θ performance, while about the same as the σ_ψ performance in the heading control case, has a much smaller influence on pilot ratings. Furthermore, because M_δ was held constant (Figs. 11b, 11d) the resulting pilot activity, σ_{δ_e}, was strongly influenced by M_u and thus emerged as tne important pilot rating factor In contrast, for the tested heading control task both σ_{δ_r} and σ_ψ appear to be equally important as pilot rating factors.

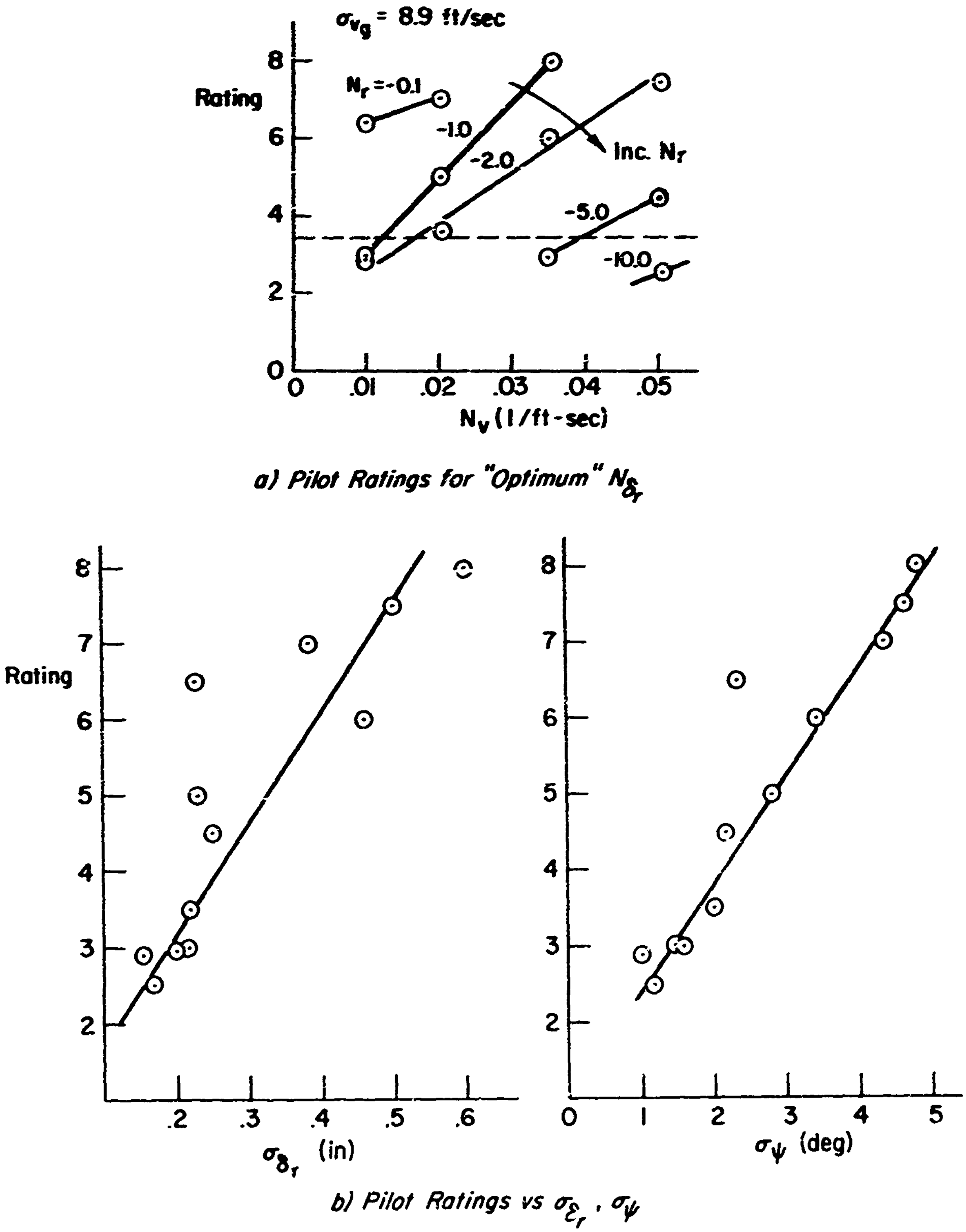


Figure 16. Correlations with Pilot Rating

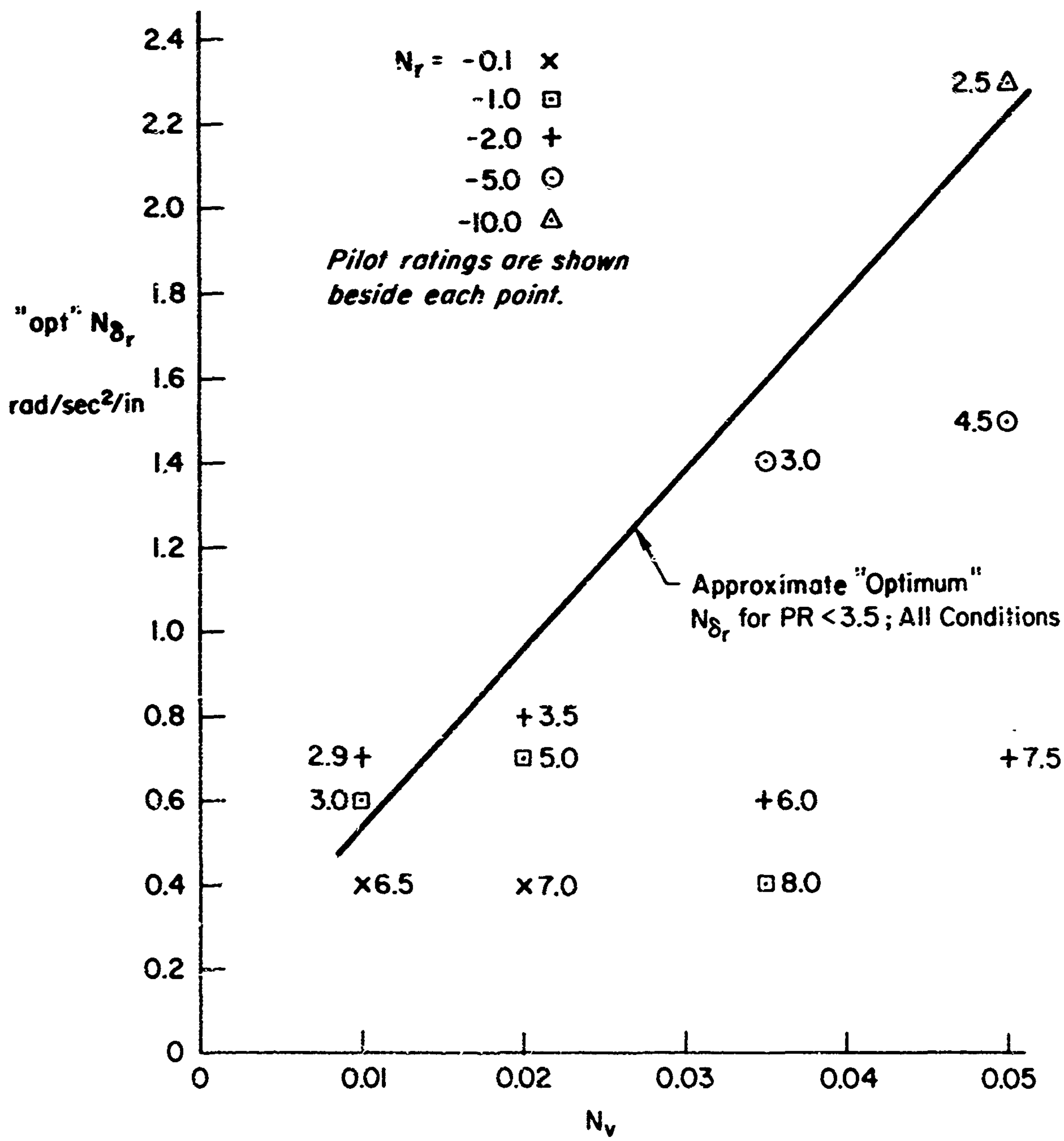


Figure 17. "Optimum" N_{δ_r} Variations with N_v and N_r

3. Summary of Important Analytical Results for Lateral-Directional Control During Approach

In tracking the ILS localizer with aileron:

a. Based on consideration of Ref. 17 results, the aircraft will have good lateral-deviation control, $y \rightarrow \delta_A$, if the heading control is good, $\psi \rightarrow \delta_A$.

b. Acceptable aileron-only heading control requires a closed outer-loop crossover frequency greater than about 0.3 rad/sec or, for the test conditions analyzed, $\zeta_\varphi \omega_\varphi$ or $1/T_{\varphi_1}$ greater than 0.4/sec (Fig. 13) in addition to minimum roll damping, $-L_p > 1.5$ sec.

For the control of gust-induced heading disturbances with "rudder" at approach speeds:

c. The pilot's rating is a strong function of the directional damping, N_r (unlike longitudinal hover), as well as the directional stability, N_v, and rudder effectiveness, N_{δ_r} (Fig. 16a).

d. The pilot's rating shows a strong correlation with rms heading response, σ_ψ, and with rms control deflection, σ_{δ_r} (Fig. 16b).

e. The optimum control sensitivity, N_{δ_r}, is a function only of N_v for a given gust input (Fig. 17).

f. The directional damping, N_r, requirements are a strong function of N_v and the design gust level, σ_{v_g}, as reflected in the value of σ_ψ (p. 38).

C. FORWARD FLIGHT LONGITUDINAL CONSIDERATIONS

Analytically, hover is an easy flight regime to study since most motions decouple into simple modes; and for pitch the dynamics are completely expressed by two to three stability derivatives which are simple to estimate. Likewise, a variety of experimental data are available from hover flight and fixed-base simulator studies. On the contrary, there is a paucity of data and information in the low speed transition range. In this region the motions become coupled and the derivatives are influenced by complex interactions which are not easy to treat analytically. This could also be said of the high speed range above 150 kt where little data exist, either analytical or experimental, in regards to handling qualities.

The transition region (hover to forward flight) is an important regime because it encompasses approach conditions and is commonly characterized

by sign reversals in either or both M_u and M_w. The industry approach has been to solve these problems with stability augmenters if the problem is serious enough or just to neglect it if the troublesome speed is low or the region of reversal small enough. However, the region and its associated handling qualities can be important if the approach speed happens to lie in it or if the helicopter has to stand plane guard off a carrier at the reversal speed.

1. Dynamical Considerations Without Gust

Fortunately the rotary damping requirements in forward flight have been found to be, for all practical purposes, the same as those determined for hover. In this regard there appear to be sufficient helicopter data (e.g., Ref. 18) which, however, do not usually include information on the other derivatives of potential importance.

For phugoid-type instabilities ($M_u < 0$) there are some non-helicopter-oriented data in Ref. 19 (Table V) for a fixed-base simulation. The cases of interest had aperiodic phugoid modes of the form $(s - 1/T_{p_1})(s + 1/T_{p_2})$ with zero total damping; i.e., $1/T_{p_1} + 1/T_{p_2} = 0$ as shown by the sketch of Fig. 18. For these situations the rating was 3-1/2 for an aperiodic "frequency" of 0.2 rad/sec ($1/T_{p_2} = -1/T_{p_1} = 0.2$). This is probably no worse than an unstable spiral mode for which the loop closure problem is very similar; and spiral divergence rates of about 4 sec to double amplitude ($1/T \doteq -0.17$) are not considered unacceptable for conditions other than cruise or landing (Ref. 48). However, from a practical standpoint the configuration is not typical of the helicopter having M_u and M_w problems in the low speed range: for example, the

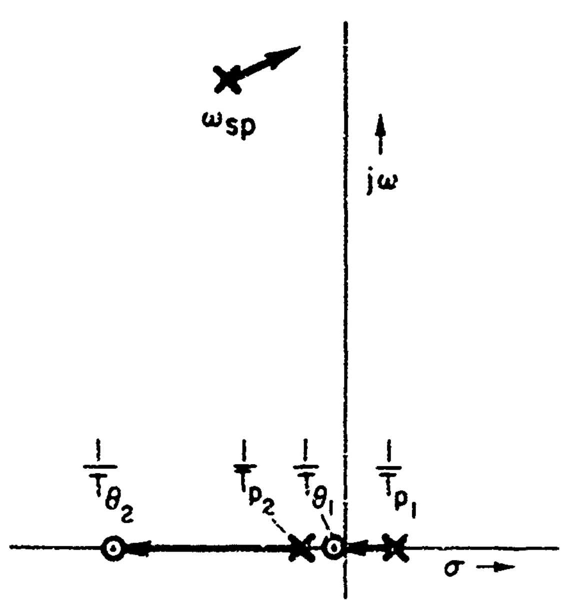


Figure 18. Phugoid Instability

separation of the short period and the phugoid is not as great in the helicopter as in the airplane.

If the short period and phugoid are separable, then the short period dynamics can be treated by the maneuver margin criterion or equivalent in terms of the "concave downward" requirement (the two have been shown to be equivalent in Ref. 20. These open-loop requirements will be discussed further in Section IV.

Several examples of possible configurations where the modes are inseparable are given in Ref. 2, and these same situations are shown not to be amenable to the concave downward requirement as discussed in Ref. 21 (p. 84). One such open-loop transfer function is depicted by the sketch of Fig. 19. For this configuration it is difficult to physically separate the modes, and the labeling in the sketch is for identifying purposes only. Airplane rating data somewhat representative of the above situation are found in Refs. 22 and 23 for negative values of maneuver margin, $\omega_{sp}^2 < 0$. The results of these data would, except for emergency situations, forbid any negative values of the aperiodic $1/T_{sp_1}$. For example, it was not possible to obtain a rating better than 4 despite high "short period" damping $(1/T_{sp_1} + 1/T_{sp_2})$. However, the short period approximations, applicable in these tests, replace the two ω_p poles and the $1/T_{\theta_1}$ zero with a single pole at the origin (free s). This is a much more difficult configuration to stabilize than those shown in either Fig. 18 or Fig. 19. In fact, if the positive $1/T_{\theta_1}$ zero shown in Fig. 19 is at least 0.1 rad/sec, the pilot closure can be made with relatively little difficulty or at least with the same ease as for the oscillatory divergent phugoid case pictured in Fig. 18.

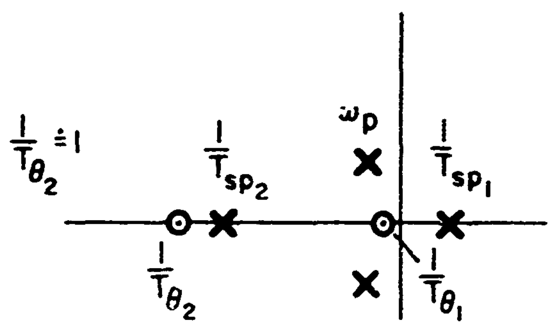


Figure 19. Typical Example of M_u and M_w Instability

Unfortunately, the number of situations similar to that of Fig. 19 (poor separation of modes), each having a problem peculiar to the specific

pole-zero locations, can be large. This makes it rather difficult to establish a general criterion to cover all situations, particularly without experimental data with which to relate to handling qualities ratings. This, then, is an area that requires further analytical-experimental effort if more realistic requirements are to be obtained.

2. Longitudinal Gust Responses in Forward Flight

This section considers the gust responses in pitch for approach conditions. Longitudinal situations are investigated for several combinations of the stability derivatives which encompass only the stable values of M_u and M_w. The pilot adaption rules are applied in order to obtain optimum performance by closing a single attitude loop, and the effects of the stability derivatives on the rms attitude response and the rms control activity are determined. Since pilot opinion data in defined gusty environments is nonexistent for the subject closures, the results presented are based only on analytical findings.

The Gust Model

The gust input spectrum is that represented by Eq. B-1. The integral scale of turbulence, L, is generally set at 1000 ft for altitudes above 1000 ft and equal to the altitude below 1000 ft. For low speed aircraft the gust break frequency is generally between 0.2 rad/sec and 1.0 rad/sec, and in the absence of flight test data the choice of ω_g has been set at 0.5 rad/sec — the same value that was found appropriate for the directional approach case of Subsection 3. This is equivalent to an average approach altitude of 225 ft as given by $\omega_g = (3/2)(V_{as}/L)$ in Eq. B-1.

Longitudinal Forward Flight

In this section the rms pitch response, σ_θ, and the rms control activity, $M_{\delta_B}\sigma_{\delta_B}$, are examined separately for the horizontal u gust (σ_{u_g}) and for the vertical w gust (σ_{w_g}) inputs. The aircraft equations are given by Eq. 1, and the following parameters are held constant at the given values:

$U_o = 75$ ft/sec	$M_q = -1.5$/sec	$Z_w = -1.0$/sec
$X_u = -0.13$/sec	$X_{\delta_B}/M_{\delta_B} = -5$ ft	$Z_{\delta_B} = 0$

The gust-sensitivity parameters, M_u and M_w were varied as follows (the M_u variations are identical to those used in the hover analysis):

$$M_u = \begin{cases} 0.0088/\text{ft-sec} \\ 0.088/\text{ft-sec} \end{cases} \qquad M_w = \begin{cases} -0.002/\text{ft-sec} \\ -0.02/\text{ft-sec} \end{cases}$$

For the $\theta \rightarrow \delta_B$ closure, the pilot model is as represented by Eq. 7, where the pilot lead is adjusted to give a phase margin of at least 30° for a crossover frequency of 2 rad/sec. The gains for closure at these conditions are called the nominal gains.

The rms gust responses in σ_θ and $M_{\delta_B}\sigma_{\delta_B}$ were computed for the horizontal σ_{u_g} and the vertical σ_{w_g} gust inputs. The purpose of these computations is to compare the relative magnitudes of these disturbances (on the pitch angle and control activity) with those obtained in hover, and also to examine the effects of changes in the M_u and M_w stability derivatives on the two responses, σ_θ and σ_{δ_B}. The rms responses were computed using the techniques outlined in Appendix B.

A comparison of the rms responses at hover with those in forward flight is given in Table I, where the gust break frequency is 1.0 rad/sec in hover compared to 0.5 rad/sec in forward flight at the approach altitude.

TABLE I

COMPARISON OF RMS RESPONSES TO u_g AND w_g GUSTS
IN HOVER AND FORWARD FLIGHT

		σ_θ (deg)		$M_{\delta_B}\sigma_{\delta_B}$ (deg/sec^2)	
		$\sigma_{u_g} = 5$ ft/sec	$\sigma_{w_g} = 5$ ft/sec	$\sigma_{u_g} = 5$ ft/sec	$\sigma_{w_g} = 5$ ft/sec
Hover	Low M_u	1.40	0	3.0	0
	High M_u	4.0	0	29.0	0
Forward flight, low M_w	Low M_u	0.55	0.085	2.6	0.40
	High M_u	5.9	0.095	23.0	0.38
Forward flight, high M_w	Low M_u	0.54	0.85	2.4	3.7
	High M_u	5.7	0.95	19.0	3.3

For all cases it appears that the control activity, $M_{\delta_B}\sigma_{\delta_B}$, in response to u gusts is greater for hover; and except for the high M_u cases in forward flight, the pitch response to u-gusts is also greater for hover. The latter exception is not serious since the survey of Fig. 1 shows that the variation of M_u with speed would place its most probable value at the low range of M_u in the above comparison. Thus the hover situation is generally worse than the forward flight condition.

The observed effects of M_u and M_w on the forward-flight rms responses to a u_g gust can be summarized as follows:

a. Increasing M_u by a factor of 10

   1) Increases σ_θ by a factor of 10
   2) Increases $M_{\delta_B}\sigma_{\delta_B}$ by a factor of approximately 8.5

b. Increasing M_w by a factor of 10 has a negligible effect on σ_θ and $M_{\delta_B}\sigma_{\delta_B}$.

Similarly, the effects of M_u and M_w on the rms responses to a w_g gust are summarized as follows:

a. Increasing M_u by a factor of 10 has a negligible effect on σ_θ and $M_{\delta_B}\sigma_{\delta_B}$

b. Increasing M_w by a factor of 10

   1) Increases σ_θ by a factor of 10
   2) Increases $M_{\delta_B}\sigma_{\delta_B}$ by a factor of 9

The computed responses to w gusts are however considered unrealistic since tight attitude control under such circumstances gives a very hard ride; that is, the rms vertical accelerations tend to approach

$$\sigma_{\Delta n_z} \doteq \frac{-Z_w \sigma_{w_g}}{g} \doteq \frac{5}{32.2} \doteq 0.16$$

The pilot more probably elects to "loosen" the θ closure (lower the pilot pitch gain) to allow natural "weathervaning" to reduce the vertical response. Under such circumstances control activity in response to w gusts is minimal and the pilot simply lets the aircraft "ride" the disturbance provided he has sufficient ground clearance. In any event, whether θ control is tight or loose, the main contributor to stick motions and attitude excursions is

due to $M_u u_g$ disturbances, except for low M_u, high M_w combinations (see Table I). For the latter, the high probability of loose control for w gusts invalidates the significance of the w gust values given in Table I.

The foregoing remarks indicate that while control responses and control margin requirements computed by pitch closure considerations are probably adequate for ordinary cruise flight, they may be inadequate for terrain-following requirements in gusty air. Very probably such requirements will be set in practice by the automatic control philosophy used to implement operational terrain-following systems. In any event, such requirements will depend on both assumed terrain and gust input characteristics.

3. Summary of Important Analytical Results from Low Speed Forward Flight Considerations

a. At low approach speed (U_o < 100 ft/sec) the open-loop longitudinal roots are not always easily identified or separable into long- and short-period modes. A possible criterion (needs experimental verification) to achieve acceptable pitch control requires that there be no unstable aperiodic roots less than about −0.2 rad/sec (p. 42).

b. Control motions (from trim) required to counteract gusts are largest at hover based on the fact that M_u is greatest at hover and that Z_w never becomes much greater than 1/sec (p. 45).

SECTION IV

OPEN-LOOP ANALYSES

A. A NOTE ON CONTROL POSITION STABILITY

There seems to be some confusion as to the meaning of "control position stability" as used in the specification (Ref. 1). Speaking precisely, the phrase refers to the change in stick position for a unit change in speed about a trim condition; i.e., it is the ratio of change in δ_B to a change in u about a trim point at U_0 for all other controls remaining fixed. This is not to be confused with the slope of the curve connecting the level flight ($\gamma = 0$) trim points of stick position versus velocity (see Fig. 20).

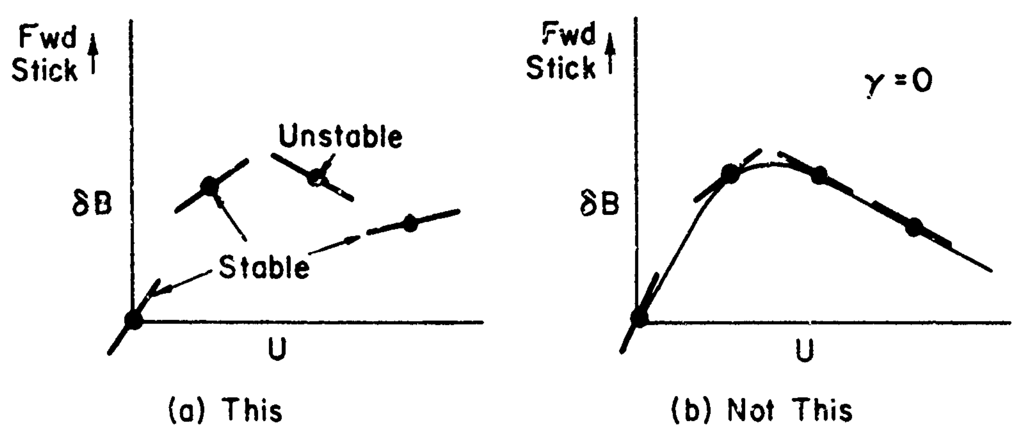


Figure 20. Control Position Stability for $\gamma = 0$

In the latter case the collective pitch is usually changed in going from one trim point to another. The specification neither refers to nor is intended to imply the use of the latter case. If the motion of the vehicle can be described by the linear equations of motion about a trim point, then the slope is the steady-state part of the transfer function of u to δ_B as given in Ref. 2:

$$\frac{\delta_B}{u} = \frac{1}{\left.\frac{u(s)}{\delta_B(s)}\right|_{(s=0)}} = \frac{E}{D_u}$$

where $E = g(Z_u M_w - M_u Z_w)$ and $D_u = g(M_{\delta_B} Z_w - Z_{\delta_B} M_w)$

For the flight ranges of the two example helicopters considered in Ref. 2, the control position stability can be represented by the approximation

$$\frac{\delta_B}{u} \doteq \frac{-M_u}{M_{\delta_B}} \tag{20}$$

Since M_{δ_B} is usually nearly constant over the speed range of the helicopter, the static stability is primarily determined by M_u variations with speed when the above relationship, Eq. 20, is valid. The condition for validity of Eq. 20 depends on the inequalities

$$Z_u M_w \ll M_u Z_w$$

$$Z_{\delta_B} M_w \ll M_{\delta_B} Z_w$$

B. THE "CONCAVE DOWNWARD" REQUIREMENT FOR MANEUVER STABILITY (ITEM 3.2.11.1 OF REF. 1)

Since the "concave downward" test was first introduced as a measure of maneuver margin, several attempts (see Refs. 20, 21, 24, 25) have been made to translate this requirement on the transient behavior of normal acceleration into more easily understood relationships involving the stability derivatives or the characteristic factors. The most noteworthy attempt is the approach presented by Seckel (Ref. 20). This approach reduces the requirement to boundaries in the "s" planes which define the corresponding regions for the characteristic short-period roots or factors (see Item 3.2.11.1 of Section V-B).

$$\begin{aligned} 2\zeta_{sp}\omega_{sp} \quad &\text{or} \quad \frac{1}{T_{sp_1}} + \frac{1}{T_{sp_2}} \doteq M_q + Z_w \\ \omega_{sp}^2 \quad &\text{or} \quad \frac{1}{T_{sp_1}} \frac{1}{T_{sp_2}} \doteq M_q Z_w - U_o M_w \end{aligned} \tag{21}$$

Although the short-period mode is fairly representative of helicopter characteristics at forward speeds, there are conditions involving negative M_u and/or positive M_α where short-period dynamics, as such, do not exist. The concave downward test in this instance can give a false indication of the handling qualities.

An example of such a situation is taken from Ref. 2 where, over the forward speed range investigated (up to 120 ft/sec), the tandem-rotor configuration possesses a positive M_w and a moderate M_u of negative sign, but with $|M_u| \ll M_w$. Under these circumstances the characteristic longitudinal roots will lie along the root locus (Fig. 21) of $G(s) + 1$,

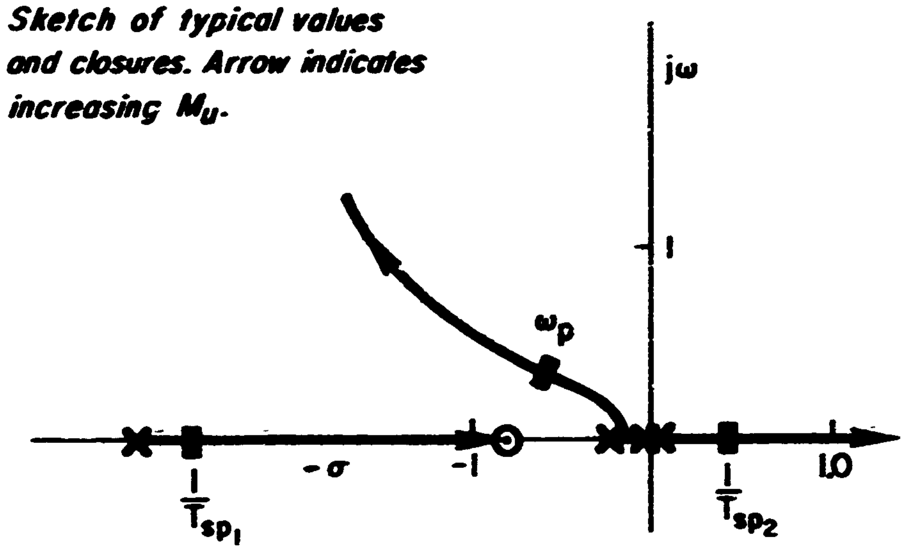


Figure 21. Intermediate Modes Arising Through Small Negative M_u and Positive M_w

where $G(s)$ is obtained from an approximation to the characteristic determinant (Ref. 2):

$$G(s) = \frac{gM_u\left(s - Z_w + Z_u\frac{M_w}{M_u}\right)}{s^2\left[s^2 + (-Z_w - M_q)s - U_0M_w + M_qZ_w\right]} \qquad (22)$$

Typical results of the closure are indicated by the ▮ symbols which become the roots of the characteristic quartic. The two short-period poles (poles are ✕ in Fig. 21) could have positive maneuver margins ($M_qZ_w - U_0M_w > 0$) and satisfy the "concave downward in 2 sec" requirement. However, undesirable handling could result due to the large divergent factor shown as $1/T_{sp_2}$ in Fig. 21. The transient response of such a configuration is similar to that shown in Fig. 22. It should be recognized that the time history depicted is not considered satisfactory

according to a complete reading of the Item 3.2.11.1 specification which requires that the response "...remain concave downward until the attainment of maximum acceleration." This portion of the specification therefore prohibits negative values of $1/T_{sp_2}$.

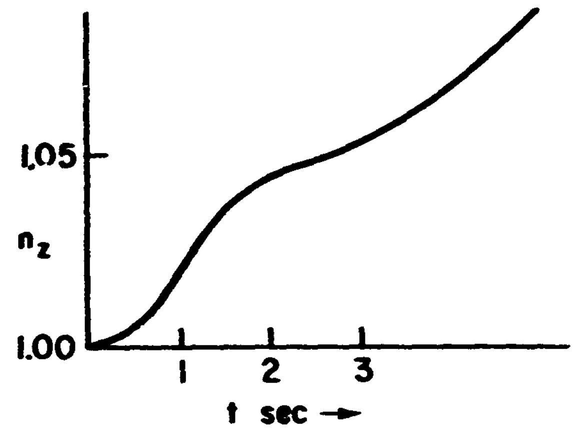


Figure 22. Transient Due to Negative M_u with Positive M_w

However, there are indications, e.g., in Ref. 19 discussed previously on p. 42, that divergent (negative) values of the aperiodic mode as high as 0.2/sec are acceptable (rated 3-1/2). At zero static margin ($U_0M_w = 0$), the minimum damping of Section III-A ($-M_q = 1.5$), and a typicale value of $-Z_w \doteq 1$ for forward flight, it is estimated that to limit the divergent factor to -0.2/sec means the equivalent of $M_u \doteq -0.0021$ at zero angle of attack stability ($U_0M_w = 0$). It appears possible, therefore, to have situations in which neither the complete "concave downward" requirement nor that for control position stability are met, but which are considered acceptable.

C. OPEN-LOOP RESPONSES TO δ_A STEP INPUTS

The roll rate response to a δ_A unit step is usually a first-order lag (assuming no dutch roll excitation) where the time constant, T_R, is approximately equal to $1/-L_p$:

$$p(t) = \mathcal{L}^{-1}\left[\frac{L_{\delta_A}}{s(s - L_p)}\right] \tag{23}$$

and the steady-state roll rate for a unit step is L_{δ_A}/L_p; the ratio of the control sensitivity to the roll damping. However, for the NASA configurations of Ref. 15 where the spiral mode ($1/T_s$) is appreciable and sometimes equal to the roll subsidence ($1/T_R$),

$$p(t) = \mathcal{L}^{-1}\left[\frac{L_{\delta_A}}{\left(s + \frac{1}{T_s}\right)\left(s + \frac{1}{T_R}\right)}\right] \tag{24}$$

In the latter case the steady-state rolling response is zero and a steady bank angle is obtained for stable spiral modes. The pilots in the NASA test of Ref. 15 gave these situations a good rating, which would seem to contradict good handling qualities requirements for conventional aircraft. The steady-bank angle of the NASA test was smoothly and quickly attained, and since heading rate is proportional to bank angle (for small sideslip), the effect was to make heading to aileron almost a pure rate system. Under these circumstances the vehicle steers like an automobile and there is no basic objection to such a response. Such responses have, in fact, been proposed as more desirable than the conventional roll-rate response, but there are other aspects which must be considered. For example, precise control of bank angle in gusty air may be more difficult — not a major problem in the NASA tests.

The computed roll, roll-rate, and sideslip time histories for the NASA and NRCC configurations of Appendix B are shown in Fig. 23. High sideslip occurs independently of the type of "p" response ($1/T_s$ large or small) and this is consistent with the pilot's complaints of high β even for the best rated configuration. Of course as speed reduces, sideslip is not a good indication of discoordination. For example, at hover a given bank angle produces a side velocity only, therefore, a 90° sideslip angle; but there is no lateral net force on the pilot who judges the maneuver to be perfectly co-ordinated. In view of such considerations, it might have been more appropriate to compute the lateral acceleration rather than sideslip angle time responses for the Fig. 23 cases.

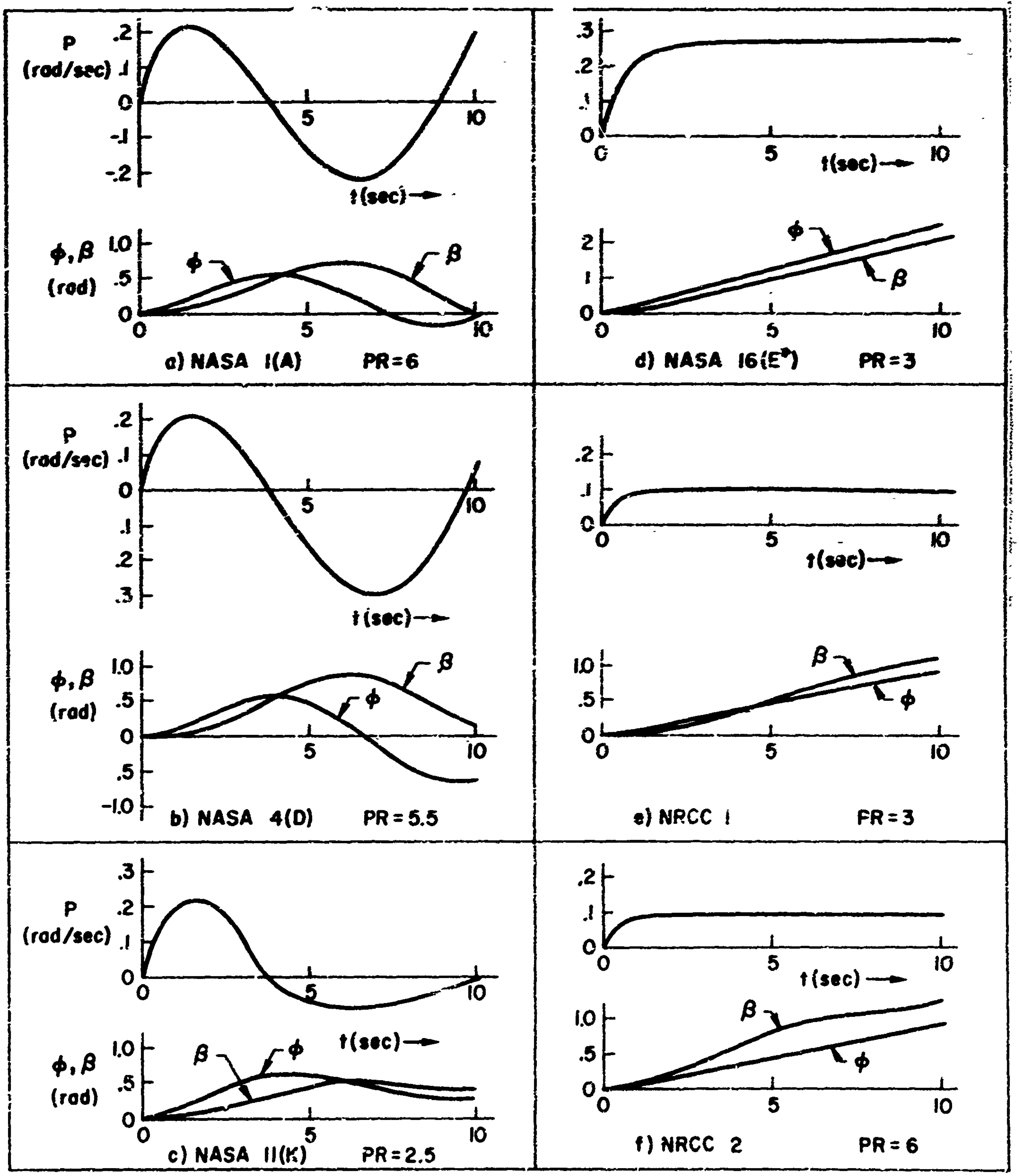


Figure 25. Open-Loop Responses to 1-Inch δ_a Step

As for the bank-angle and roll-rate time histories, there is a clear distinction between the right and left columns as regards the type of response. Obviously the right set represents roll-rate control in accordance with Eq. 23, whereas the left set, governed by Eq. 24, shows examples of bank angle proportional to input. However, if the steady bank angle is achieved reasonably fast and has small residual oscillations (Fig. 23c), the pilot ratings are about the same as for the "ideal" roll-rate cases. Notice that such good ratings are delivered despite the obvious roll-rate reversals that must occur to limit the bank angle. Such reversals, clearly an integral part of the maneuver, cannot be bad. On the other hand, considering roll-rate control, reversals that can occur due to dutch roll interactions excited by $\omega_\varphi/\omega_d \neq 1$ (Ref. 4) are undesirable.

The cause of the large difference in pilot rating between Figs. 23d, 23e, and 23f is not evident from the time traces shown. However, if we consider heading response (not depicted) to be given by (for $Y_v \doteq 0$, as was the case) $r = (g/U_0)\varphi - \dot{\beta}$, then it is clear that the characteristics corresponding to Fig. 23f are much worse than those of Figs. 23d and 23e. For example (for $U_0 \doteq 50$ ft/sec), it appears that for the former case the heading does not respond in the desired direction for about the first 3 sec. This poor heading response carries over, of course, into the closed-loop situation already discussed in connection with Fig. 13, where the comparisons clearly show the inferiority of the NRCC 2 configuration.

D. SENSITIVITY AND CONTROL POWER

Generally speaking, it has been very difficult to obtain any sort of handle on sensitivity or control power requirements either by analytical means or from experimental data sources. For the closed-loop analyses of Section III the sensitivity was always assumed optimum and the control power required for the closed-loop tasks was always found to be a small percentage of the total power usually available.

There appear to be many examples in the literature, Section V, where the specification requirements are considered inadequate and most references seem to feel that control sensitivity and power should be specified as a function of mission instead of weight, although there is very little

information as to what the mission or maneuver requirements should be in terms of either control sensitivity or power.

1. Longitudinal Control Sensitivity

From the closed-loop analyses of Section III-A there is some evidence that the optimum longitudinal control sensitivity has the form of Eq. 17: a basic value plus a term dependent on the gust sensitivity. Experimental evidence from Ref. 14 indicates that for the longitudinal hover case the optimum sensitivity, M_δ, is dependent on M_u and M_q, whereas in the directional approach case of Ref. 16 the optimum N_δ sensitivity for gust disturbances appeared almost directly proportional to N_v (see Fig. 16b).

The term independent of M_u in Eq. 17 is best "explained" from open-loop considerations. Taking "optimum" M_δ versus M_q data from longitudinal examples for $M_u = 0$ show these data to correlate with an angular response in 1 sec time. This correlation is given in the presentation of Fig. 24 where the optimum sensitivities are replotted on the $M_{\delta_B}\delta_B$ versus T_{sp} ($\doteq 1/M_q$) grid (see Ref. 3) for δ_B = 1 inch. For unit stick step inputs the ordinate

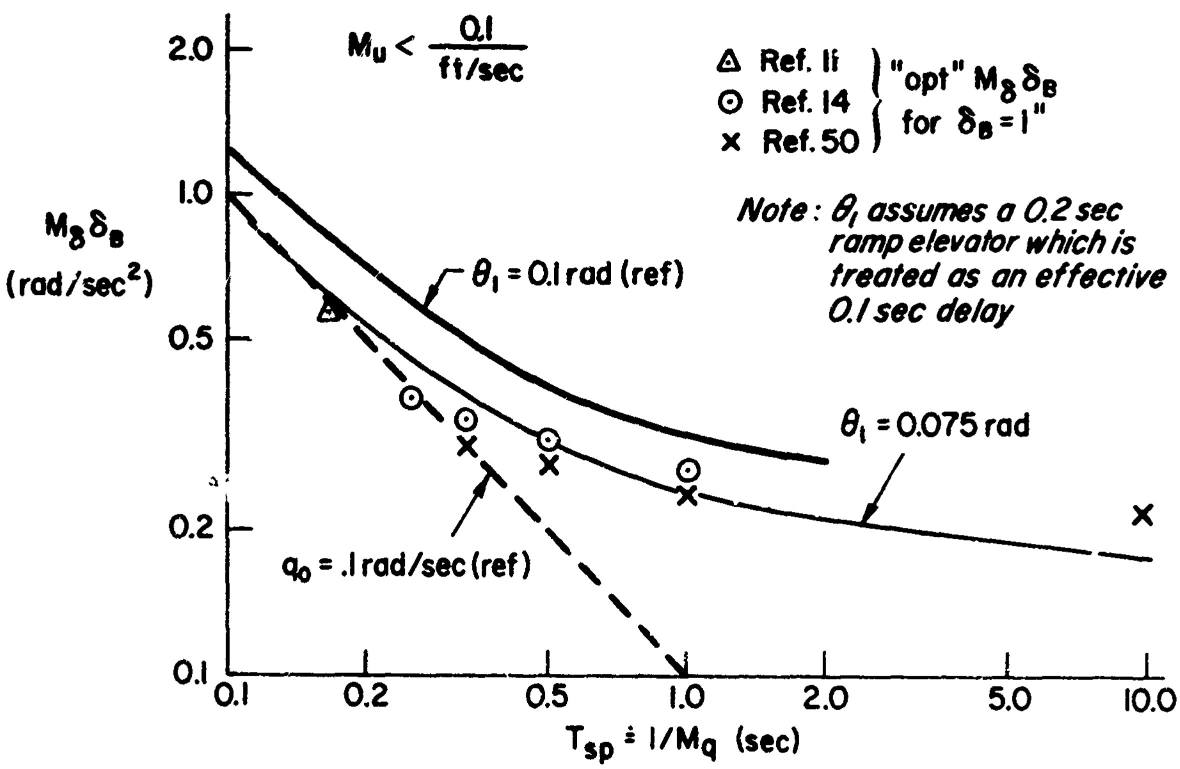


Figure 24. Optimum Longitudinal Control Sensitivity for Low M_u

becomes the optimum control gain, M_{δ_B}. Lines of constant pitch angle in 1 sec, θ_1, and steady pitch rate, q_o, are parallel to the heavy reference lines shown. We see that $\theta_1 = 0.075$ rad $= 4.3^\circ$ represents the kind of pitch response the pilots find most desirable for these configurations. The points in Fig. 24 represent data obtained from both flight test and fixed-base simulator for best-opinion M_δ where M_u is known to be small. On this basis the optimum gain in still air is set by the damping level to give a desired attitude displacement in 1 sec. This means that the minimum control sensitivity for $M_u = 0$ and for the minimum acceptable damping of 1.5 ($T_{sp} = -1/M_q = 0.67$) is approximately 0.25 (rad/sec^2)/inch from Fig. 24. An additional ΔM_δ above this basic value is required for gust sensitivity (M_u) effects and possibly for extra mission requirements. However, because M_u is unknown for much of the longitudinal data on M_q versus M_δ, it is hard to obtain enough data to establish the "additional" requirement.

2. Directional Control Sensitivity

The results of Ref. 16, already discussed in the preceding section in connection with the heading control task, also seem to support a response in 1 sec criterion. For example, the data points lying along the "over-all optimum" line of Fig. 17 are shown in Fig. 25 on a grid similar to that of Fig. 24. It may be seen that the "over-all optimum" N_{δ_r} versus N_r points correspond closely to $\psi_1 \doteq 0.19$ rad $\doteq 10.9^\circ$. Of course the actual responses are somewhat dependent also on the value of U_oN_v, which varies with N_{δ_r} (Fig. 17), but is not included in the simplified calculations on which Fig. 25 is based. However, the errors involved in neglecting this effect are small (less than about 10 percent) for the tested values of N_v ($U_o \doteq 50$ ft/sec) and are probably less than the uncertainties involved in identifying the optimum N_{δ_r}'s.

We must remember that these results were obtained in simulated gustiness corresponding to $\sigma_{v_g} = 8.9$ ft/sec and that the responses are probably greater than those that would be considered optimum for a nongusting environment. This follows from the notion, explored above (p. 38), that the pilot-selected values of "optimum" N_{δ_r} reflect his desire to reduce pedal activity, in the

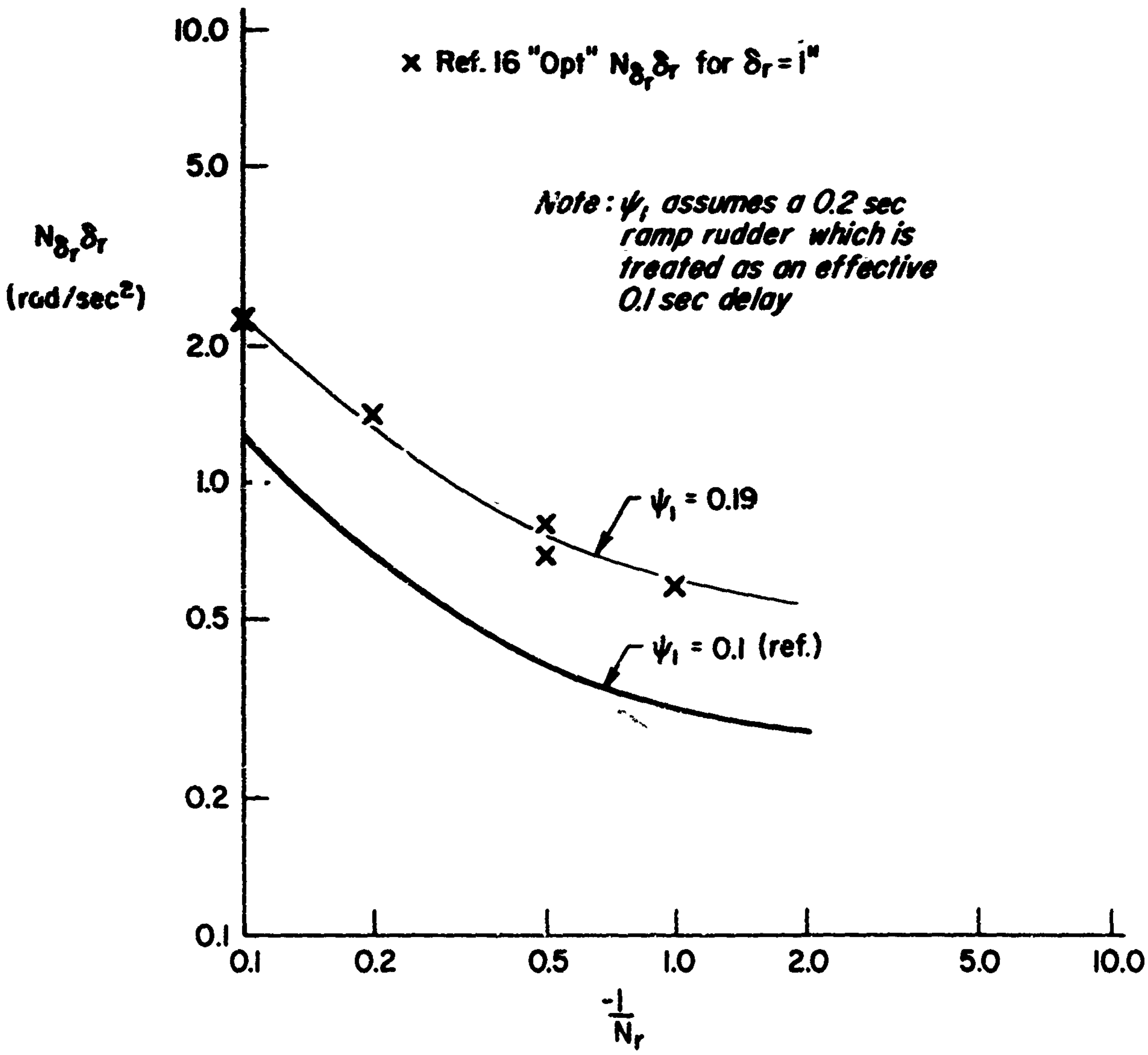


Figure 25. Optimum Directional Control Sensitivity

presence of gusts, to an acceptable level. Incidentally, the over-all optimum N_{δ_r} values shown in Fig. 17 and plotted in Fig. 25 are essentially the same for either the hover or circuit-flying ($U_0 < 40$ kt) tasks of Ref. 16.

3. Lateral Control Sensitivity

Figure 26, taken from Ref. 3, shows the variation of optimum roll power for a number of lightweight (2500—4500 lb) flight-tested VTOL configurations. As noted in Ref. 3, these data are probably more indicative of desirable control sensitivity than of control power because the control actions used by the pilots were all in the small deflection range. For a maximum stick displacement of 5 inches (as in the X-14) the optimum value of φ_1 per inch is about 0.15 rad $\doteq 8.6^\circ$.

4. Control "Harmony"

If we discount slightly the values of ψ_1, noted above, because of the quite gusty environment in which they were obtained, we see that, roughly, the optimum control sensitivities are in the ratio

$$\theta_1 : \psi_1 : \varphi_1 \doteq 1 : 2 : 2$$

We can compare these with the ratios specified by MIL-H-8501A which are given as

$$\theta_1 : \psi_1 : \varphi_{1/2} = 45 : 110 : 27$$

To effect a direct comparison we convert $\varphi_{1/2}$, the bank angle attained in 1/2 sec, to φ_1 by multiplying by a factor of 4. (For the 0.1 sec effective time delay assumed, the actual factor varies from 3.9 at $T_R = 0.5$ to 4.4 at $T_R = 1.0$.) Then the MIL-H-8501A-specified ratios are approximately

$$\theta_1 : \psi_1 : \varphi_1 \doteq 1 : 2.45 : 2.4$$

These values are in reasonable agreement with those based on the data presented above.

On the other hand, simulator experiments, Refs. 27 and 49, tend to show that the desired 1 sec response about the various axes is about equal if one takes into account the speed stability associated with the axis (M_u, L_v, and N_v). However, flight test results tend to agree with the preceding sensitivity ratios, possibly because the pilot has other criteria which

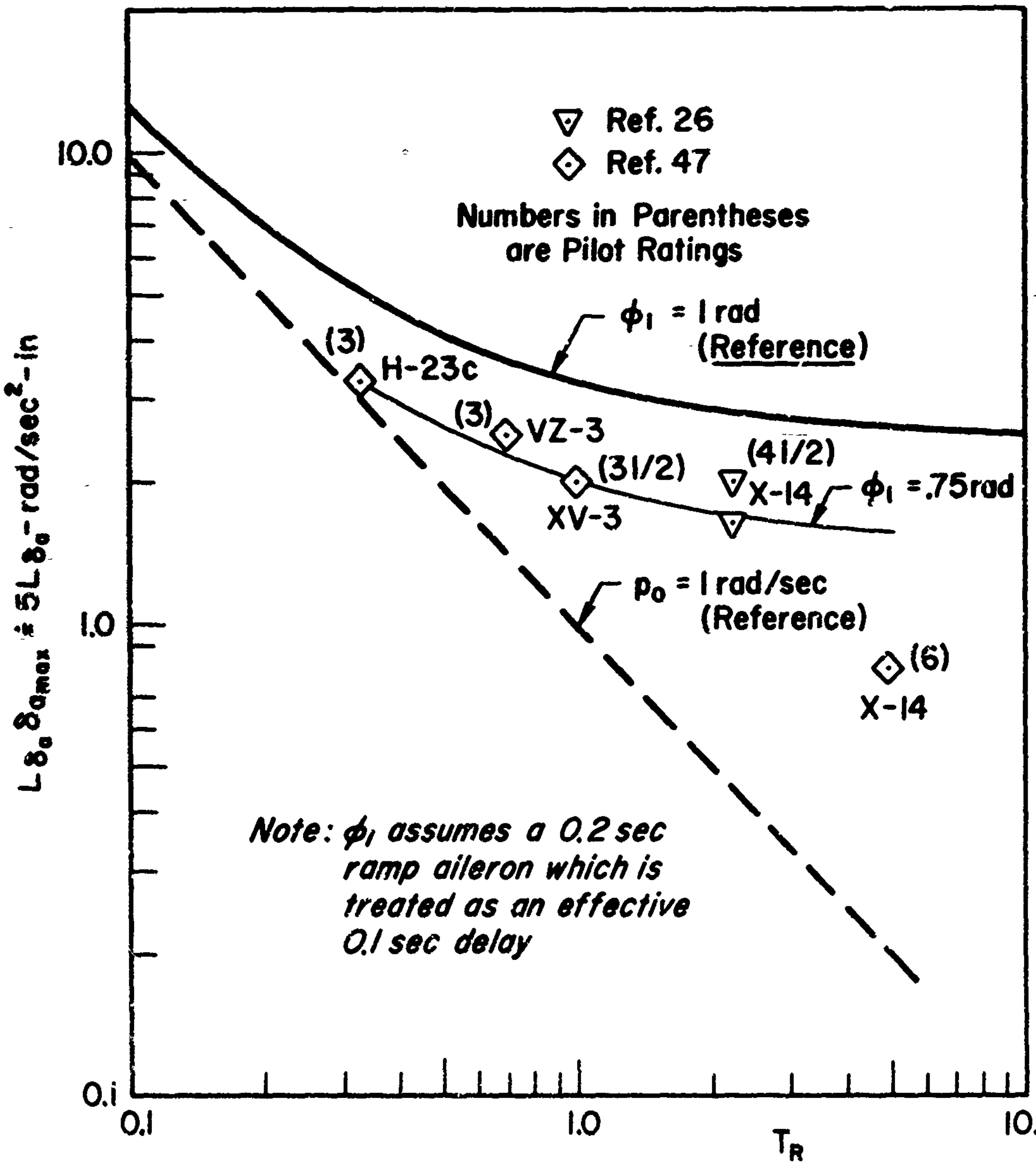


Figure 26. Optimum Lateral Control Power

lead to a different sensitivity between the longitudinal and the lateral-directional axes. One such criterion could be associated with maintaining zero side velocity on the landing gear.

As regards the absolute values of the desired response in 1 sec, more data are required to firmly establish the general validity of the above numbers, especially with respect to the possible influence of size or mission. In terms of the above discussion, the question of size effects boils down to whether pilots will accept degraded 1 sec performance as size

increases. There is little in the above data to reject or verify such a possibility.

E. SUMMARY OF IMPORTANT ANALYTICAL RESULTS FROM OPEN-LOOP CONSIDERATIONS

1. The static longitudinal stick stability with speed is given (p. 49) for any trim speed by the approximate expression

$$\frac{\delta_B}{u} \doteq \frac{-M_u}{M_{\delta_B}}$$

whose validity depends mostly on the inequality

$$Z_u M_w \ll M_u Z_w$$

2. The "concave downward" requirement of the specification may be unduly restrictive (p. 51), considering tolerable values of negative M_u (approximately −0.002).

3. These negative values of M_u can result in unstable variations of stick position with speed which, again, appear to be tolerable.

4. For helicopters with low N_v and a large spiral mode, lateral stick inputs command bank angle and associated heading rates rather than roll rates (pp. 53 and 54), but such characteristics can be acceptable.

5. The desirable level of control sensitivity in still air conditions and at the minimum required damping, 1.0 to 1.5/sec, is approximately 0.25 rad/sec^2/inch for the pitch axis, reflecting a θ_1 of about 4.5°, and about twice this value for the yaw and roll axes (p. 56). Additional control sensitivity is required for gusty air, depending on the velocity stability.

SECTION V

CRITIQUE OF SPECIFICATION

This section is an evaluation of the specification in light of the results which were obtained from the open- and closed-loop analyses of the preceding sections. These analyses form the basis for the selection of criteria which can be used to make meaningful correlations among the pertinent experimental data. In many instances the correlations, while good, are of limited impact on a specification because they are of limited scope despite the great amount of extant handling qualities experimental data. Unfortunately, the great majority of such documented experiments are incomplete and are not useful in the approach taken in this study because they do not exhibit the following:

- A description of the mathematical model (i.e., the stability derivatives)
- Carefully controlled test environment, such as gust and purity of task
- Reliable pilot ratings for each individual task

These three requirements eliminate almost all of the early work on helicopter handling qualities before the late fifties (pre-Cooper rating) as candidates for the types of correlations presented in this report. The best sources of data were the variable-stability helicopter tests conducted by Princeton (HUP-1), NASA-Langley (Vertol 107), and the Canadian Research Council (Bell H-13).

Despite this relatively weak experimental base, it is possible to pinpoint consistencies and inconsistencies, and to translate these into tentative recommendations for acceptance, rejection, or modification of a spec item. In this process, and because of the limited amount of complete correlatable data, as discussed above, we also consider other data, experiences, etc., as pertinent inputs to the specification critique. This is consistent with our standard practice of checking all applicable experience, whether directly analyzable or not, against the implications of analytically based correlations. Most of such "incidental" experience used to guide the

critique has already be-. cited in the foregoing sections. However, because many of the spec items are of a detailed nature, not amenable to general discussions, some of the "incidental" experience is cited in situ. This is especially true of comments and ideas expressed during a survey conducted by the authors and documented in Appendix C.

A. COLLECTED SPECIFIC RESULTS OF EXPERIMENTAL/ANALYTICAL CORRELATIONS

The summary statements previously given in Sections III and IV are collected here for easy reference, as follows:

1. To reduce the pilot's lead requ.rements to where they have a negligible effect on rating, the minimum damping about all axes in the absence of gust should be greater than 1.0/sec (p. 12). For multiloop control such as attitude stabilization and hovering over a spot, the minimum damping should be increased to 1.5/sec (p. 21).

2. For pitch control in still air, M_u is relatively unimportant from a dynamic standpoint; increasing M_u improves the ability of the helicopter to hover over a spot (still air only) by providing a larger crossover frequency in the position closure (pp. 18, 21, 24).

3. Increasing the numerator factor $1/T_{\theta_1}$ through nonaerodynamic augmentation improves the precision hover performance, σ_x, in both still and gusty air (p. 23).

4. Selecting θ and x pilot-closure gains according to established "adjustment" rules provides a good composite minimum to σ_θ, σ_x, and σ_{δ_B} responses in gusty air (the same is true for the lateral case in hover) (p. 27).

5. In gusty air the rms control response, σ_δ, is roughly proportional to $M_u \sigma_{u_g}$ (p. 29).

6. An increase in M_q above the minimum of Item 1 produces small favorable changes in attitude responses, σ_θ, and control deflection, σ_δ, in gusty air (p. 29).

7. Correlation of pilot rating data in hovering over a spot with analytical results of the attitude and position manual closures indicates that pilot rating is primarily a function of control activity, σ_δ, provided the Item 1 condition is met (p. 29).

8. Analysis of recent experiments shows the optimum control sensitivity, M_δ, to be a function of M_u, as well as M_q, in the presence of gusts. The optimum M_δ is a simple function of both damping, M_q, and the velocity stability, M_u (p. 30).

9. The control power required for the maximum gust expected can be specified on the basis of the rms control deflections for random gust inputs; for example (p. 29),

$$\delta_{max} = 4\sigma_g$$

10. All the foregoing apply also to lateral and directional control in hover with appropriate modifications in the motion quantities and derivatives, i.e.,

$$M_q \rightarrow L_p \rightarrow N_r \qquad M_\delta \rightarrow L_\delta \rightarrow N_\delta$$

$$M_u \rightarrow -L_v \rightarrow N_v \qquad \theta \rightarrow \varphi \rightarrow \psi$$

$$x \rightarrow y$$

11. In tracking the ILS localizer with aileron, the aircraft will have good lateral-deviation control, $y \rightarrow \delta_A$ if the heading control is good, $\psi \rightarrow \delta_A$ (p. 32).

12. Acceptable aileron-only heading control requires a closed outer-loop crossover frequency greater than about 0.3 rad/sec, or, for the test conditions analyzed, $\zeta_\varphi \omega_\varphi$ or $1/T_{\varphi_1}$ greater than 0.4/sec in addition to minimum roll damping, $-L_p > 1.5$, (Fig. 13).

13. For the control of gust-induced heading disturbances with "rudder" at approach speeds, the pilot's rating is a strong function of the directional damping, $-N_r$ (unlike longitudinal hover) as well as the directional stability, N_v, and rudder effectiveness, N_{δ_r} (Fig. 16a).

14. In the Item 13 task the pilot's rating shows a strong correlation with rms heading response, σ_ψ, and with rms control deflection, σ_{δ_r} (Fig. 16b).

15. For the approach task of Item 13 the optimum control sensitivity, N_{δ_r}, is a function only of N_v for a given gust input (Fig. 17).

16. The directional damping, N_r, requirements, based on the control task of Item 13, are a strong function of N_v and the design gust level, σ_{v_g}, as reflected in the value of σ_ψ (p. 38).

17. At low approach speed (U_o < 100 ft/sec) the open-loop longitudinal roots are not always easily identified or separable into long- and short-period modes. A possible criterion (needs experimental verification) to achieve good pitch control requires that there be no unstable aperiodic roots less than −0.2 (p. 42).

18. Control motions (from trim) required to counteract gusts are largest at hover, based on the fact that M_u is greatest at hover and that Z_w never becomes much greater than 1/sec (p. 45).

19. **The static longitudinal stick stability with speed is given (p. 49) for any trim speed by the approximate expression**

$$\frac{\delta_B}{u} \doteq \frac{-M_u}{M_{\delta_B}}$$

whose validity depends mostly on the inequality

$$Z_u M_w \ll M_u Z_w$$

20. **The "concave downward" requirement of the specification may be unduly restrictive (p. 50), considering tolerable values of negative M_u (approximately −0.002).**

21. **These negative values of M_u can result in unstable variations of stick position with speed, which, again, appear to be tolerable.**

22. **For helicopters with low N_v and a large spiral mode, lateral stick inputs command bank angle and associated heading rates rather than roll rates (pp. 53 and 54), but such characteristics can be acceptable.**

23. **The desirable level of control sensitivity in still air conditions and at the minimum required damping, 1.0 to 1.5/sec, is approximately 0.25 rad/sec^2/inch for the pitch axis, reflecting a θ_1 of about 4.5°, and about twice this value for the yaw and roll axes (p. 56). Additional control sensitivity is required for gusty air, depending on the velocity stability.**

B. SPECIFICATION-RELATED CONCLUSIONS

Before delving into the individual specification items, it is expedient and desirable to summarize and express the foregoing results in a way that relates them more directly to the specification, as follows:

1. **Rotary damping** about all axes should be a minimum of about −1.0 to −1.5/sec. Such values appear necessary to avoid degraded opinion due to excessive pilot lead generation.

Applicable to the following MIL-H-8501A items (see pp. 80–109):

3.2.11	3.3.5	3.3.15
3.2.13	3.3.6	3.3.18
3.2.14	3.3.7	3.3.19

2. Control sensitivity

a. Minimum "optimum" values appear to be associated with the maneuver response criteria expressed in terms of the angular displacement achievable in 1 sec. For a 1 inch step control displacement, "optimum" values are $\theta_1 \doteq 4.5^{\circ}$ and about twice this value for φ_1 and ψ_1.

b. The minimum values of control effectiveness (rad/sec^2-inch), corresponding to Items 1 and 2a, may result in relatively large rms control inputs, σ_{δ}, and position errors, σ_{ε}, if the gust excitation ($M_u u_g$, $L_v v_g$, $N_v v_g$, etc.) is sufficiently high. In general, an additional increment in control effectiveness proportional to the gust sensitivity derivative (M_u, L_v, N_v, etc.) is required to limit control activity and/or attitude or translation errors to acceptable values. Where attitude rather than translation errors are of primary importance, increased rotary damping above the level specified in Item 1 above will have a small beneficial effect.

c. If the additional required effectiveness due to Item b above results in appreciably increased 1 sec responses, the rotary damping must be increased to reduce the now-oversensitive response to no more than about 150 percent* of the nominal Item a values.

Applicable to the following MIL-H-8501A items (see pp. 72–106):

3.2.2	3.3.3	3.3.7	3.3.18
3.2.13	3.3.5	3.3.15	

3. **Control power** must in general be sufficient to:

a. Trim the aircraft about all axes over the desired normal and and emergency operating conditions, including atmospheric winds.

b. Maintain or recover control in gusty air

c. Perform required maneuvers consistent with the aircraft's effective utilization

The Item 3c requirement demands consideration of the aircraft's specific mission and intended use. Such considerations are beyond the scope of this study, but they usually seem to involve normal force and performance

*A rough estimate based on data in Ref. 18; also supported by Fig. 25, which shows optimum ψ_1 relatively invariant for various N_v's.

capacity rather than control power, per se. That is, generally speaking, the time spent in maneuvering by pulling load factor or accelerating, climbing, etc., is usually much longer than that spent in changing attitude (e.g., Ref. 8).* Since the foregoing Item 2 sensitivities are pilot-selected to give him adequate response and control for relatively small control inputs (recall the relatively small σ_δ's shown in the Section II analyses), it seems likely that maximum maneuver response requirements will be less than those obtainable by linearly extrapolating the desired sensitivities to full throw. A suitable factor may be something of the order of 60 percent of full throw, corresponding to about 3 inches of stick (lateral and longitudinal) and about 2 inches of rudder. Applying such deflections to the Item 2 sensitivities results in maneuvering capabilities of:

$$\theta_1 \doteq 13.5^\circ$$
$$\varphi_1 = 27^\circ$$
$$\psi_1 = 18^\circ$$

Whether such figures would be generally applicable to all helicopters is of course very unlikely. The real maneuvering requirement must be a function of the vehicle's mission and expected use. For example, $\psi_1 = 18^\circ$ is obviously an inadequate maneuver capability for an armed helicopter requiring quick turnaround at low speeds. On the other hand, $\varphi_1 = 27^\circ$ may be overstating the roll maneuverability requirement for very large helicopters, since a corresponding requirement for large airplanes in the approach condition is less than about 6° (Ref. 8).

Once maneuvering requirements are established on the basis of the helicopter's mission, etc., it would then be possible to add to these the requirements imposed by considerations of trim and gusts. Such superposition to be correct should employ a suitable probabilistic model encompassing maneuvers and gusts and their interactions. Of course the specific gust sensitivity terms (M_u, L_v, N_v, etc.) of the individual designs would govern the magnitude of such additional requirements, as they apparently also do for the desired control sensitivity (i.e., Items 2b and 2c).

*Quick turnaround in hover is an obvious exception.

To the extent that missions, expected usage, and governing gust-sensitive derivatives are a function of size and geometric configuration, logical control power requirements must also reflect such dependence. To lump such dependence into the factor $\sqrt[3]{W+1000}$ as in MIL-H-8501A seems a gross oversimplification. More research in the area of mission effects on maneuvers, and in the proper superposition of the individual control power components, Items a, b, and c above, is needed.

Applicable to the following MIL-H-8501A items (see pp. 88–106):

3.2.13	3.3.1	3.3.3	3.3.5	3.3.8
	3.3.2	3.3.4	3.3.6	3.3.18

4. **Heading control with ailerons in approach.** Good flight path control during approach appears to be synonymous with good heading control <u>without</u> the use of rudders. Furthermore, this translates into a requirement for a reasonable bandwidth, ≥ 0.3 rad/sec, for closed-loop control of heading with aileron, using a closed inner loop of bank angle to aileron. In general there are a variety of bank angle and heading transfer function <u>numerator</u> characteristics (as well as the usual denominator dynamics) which enter into the complete picture. However the data analyzed in Section III minimize heading numerator effects because the associated zeros were always favorably located. The results of the analyses were therefore compactly expressible in terms of requirements on only the φ numerator, $\zeta_\varphi\omega_\varphi$ or $1/T_{\varphi_1}$.

The open-loop consequences of the closed-loop requirement need more complete analysis before they can be stated in simple flight-test-producible terms. For example, it may be possible to judge control adequacy by considering, as with current requirements on conventional airplanes, the sideslip excursions following a step aileron input. However, in this case, both the magnitude of the sideslip and its phasing with respect to the input must be taken into account. Furthermore, a requirement expressed in such terms provides, at best, an <u>indirect</u> indication of the apparent piloting problem. A more <u>direct</u> approach may be to consider the nature of the heading time response and, for example, to specify limits on the time required to achieve initial heading rate in the correct direction. More work in this area, encompassing a wider variety of heading and roll transfer function characteristics, is needed before desirable open-loop properties can be formulated. In the

meantime suitable closed-loop properties have been identified and can be used, at least analytically.

Applicable to the following MIL-H-8501A items (see pp. 98–100):

3.3.8 3.3.9.1 3.3.9.2

5. **Roll dynamic characteristics** are usually such to produce a steady roll-rate in response to a step aileron input. However, for low speed flight where the spiral mode, $1/T_{s_1}$, can be large, the steady-state response in such cases approaches a constant bank angle. These situations are apparently acceptable from the standpoint of roll response, provided the steady bank angle is smoothly and quickly attained. In effect, the vehicle steers like an automobile, requiring a constant wheel or stick deflection to hold a steady turn rate (proportional to bank angle). When considered for conventional airplane roll-control systems, such characteristics are sometimes downrated because the pilot becomes unhappy about holding an aileron deflection throughout a long high speed turn. However, for low speed flying the turn rate is high ($r \doteq g\varphi/U_0$) and the time spent in turns is sufficiently short to apparently alleviate this complaint.

Applicable to: 3.3.9.1

6. **Longitudinal maneuvering and speed stability characteristics.** As discussed in Section IV, the "concave downward" requirement, Item 3.2.11.1, defines acceptable maneuvering characteristics to exclude mildly divergent aperiodic modes. However, there is some evidence (p. 42) that such modes, having time constants greater than about 5 sec, are acceptable. Unfortunately there is little definitive data on this point and the question needs experimental resolution. Furthermore, since such unstable aperiodicities will in most practical instances be associated with negative M_u and/or positive M_α, the resulting control position variation with speed (Eq. 20) will be slightly unstable. Similar instabilities, incidentally, are rather common on conventional propeller-driven aircraft in high power climb, where slipstream effects and direct forces on the propellers can combine to give unstable M_u's sufficiently powerful to overcome fairly respectable static margins. Provided control force variations with speed are stable, the unstable control position variations resulting from such effects are not

normally considered unacceptable; whether they would be for extended operating times (e.g., cruise) is a moot point.

The entire question of allowable negative M_u, considering both dynamic and static aspects, needs further research.

Applicable to the following MIL-H-8501A items (see pp. 78-86):

3.2.10   3.2.11.1   3.2.11.2

C. DETAILED REVIEW OF SPECIFICATION MIL-H-8501A

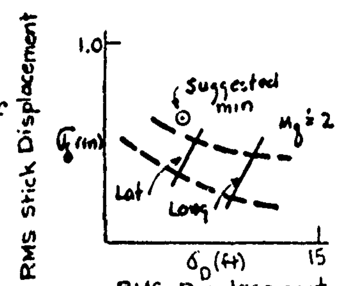
The above conclusions, based on the specific analyses and correlations conducted during the present study, reveal certain inadequacies in the applicable specification items—and in the background state of art. These inadequacies can be translated, almost by direct inference, into a set of recommendations. However, there is a considerable amount of applicable, but so far uncorrelated, data and experience which was also reviewed and which has some bearing on the final recommendations. The format used to compactly encompass all the considerations involved is to present them in tables which contain the following columns:

Specification Item: The review is confined to those specification items associated with the fundamental stability and control aspects of the helicopter and does not cover such items as vibration, instrument flight, stability augmentation, etc.

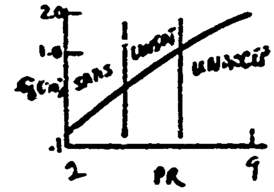
Applicable Literature: The results and conclusions of the pertinent published material are listed.

Comments: This column contains comments on conflicts between the specification item and the applicable literature and within the literature itself. Also, the results of the Government and Industry Survey (Appendix C) are cited here when applicable.

Discussion and Recommendations: A discussion of the reasonableness of the specification item based on the foregoing and on the specific conclusions listed in Section V-A and V-B above, with recommendations for acceptance, rejection, modification, or additional research.

SPECIFICATION ITEM	APPLICABLE LITERATURE
1. SCOPE **1.1** This specification covers the design requirements for flying and ground handling qualities of U.S. military helicopters. **2. APPLICABLE DOCUMENTS** There are no applicable documents. **3. REQUIREMENTS** **3.1 General.** **3.1.1** With the exception of 3.6, section 3 contains the requirements for the flying qualities, and for certain relevant ground-handling characteristics, of all helicopters procured by the Department of the Army, the Department of the Navy, and the Department of the Air Force, that are required to operate under visual flight conditions. Paragraph 3.6 applies to helicopters required to operate under instrument flight conditions. The required characteristics are those which are considered, on the basis of present knowledge, as tending to insure satisfactory handling qualities and are subject to modification as indicated by new information. Every effort shall be made by designers to provide additional desirable characteristics which have been omitted as specific requirements. **3.1.2** Unless otherwise specified, the requirements of section 3 shall apply at all normal service loadings over the operating rotor speed range and all operational altitudes and temperatures. For the purposes of section 3, normal service loadings shall include all combinations of gross weight and center of gravity locations that could ordinarily be encountered in normal service operations. **3.2 Longitudinal characteristics.**	
CONTROL POWER IN UNACCELERATED FLIGHT	**KLINAR and CRAIG (Ref. 13), IAS PAPER 61-60**
3.2.1 It shall be possible to obtain steady, smooth flight over a speed range from at least 30 knots rearward to maximum forward speed as limited either by power available or by roughness due to blade aerodynamic limitations, but not by control power. This speed range shall be construed to include hovering and any other steady state flight condition, including steady climbs and steady descents. Throughout the specified speed range a sufficient margin of control power, and at least adequate control to produce 10 percent of the	One- to three-axis fixed-base simulator. Representative of 35,000 lb duct-fan VTOL. The study emphasized measured pilot's stick activity and performance (position error) for IFR hover in still and gusty air. The standard deviations of the fluctuating velocity were a percent of the mean wind, as listed (p. 72). 

COMMENTS	DISCUSSION AND RECOMMENDATIONS
The Naval Air Test Center, Patuxent River, Md., spends a lot of time trying to determine "normal service loadings." Industry is concerned over this general operational clause because the aircraft are not tested to these conditions under the demonstration specs. Satisfying these altitude and temperature requirements could compromise over-all design.	This item is apparently too broadly written to be useful in practical handling qualities demonstrations, although its <u>intent</u>, to provide acceptable characteristics for all "normal" flight conditions, is clear. <u>Recommendation</u>: The conditions for which handling qualities are to be <u>demonstrated</u> should be more clearly delineated, perhaps by reference to type specs or to a demonstration spec.
The cited literature (IAS 61-60) does not provide a deterministic means of obtaining the control power requirements for gusty air, i.e., for the graph shown, σ_δ versus σ_D, the important factor, M_u, is missing. Also, the vertical gust component is on the low side for approach altitudes over 50 ft.	The applicable analytical results, summarized in Paragraphs 3, 4, 5, 7, 11, and 18 of Section V-A, show that pilot rating is proportional to stick activity, in turn proportional to the pitching moment disturbance level, $M_u\sigma_{u_g}$; and that minimum longitudinal power required for control of σ_{u_g} gusts (not including pilot remnant) is approximately given by $M_{\delta_B}\sigma_{\delta_B} \doteq 1.4\, M_u\sigma_{u_g} \frac{rad/sec^2}{ft/sec}$

SPECIFICATION ITEM	APPLICABLE LITERATURE
3.2.1.....................Continued maximum attainable pitching moment in hovering shall be available at each end to control the effects of longitudinal disturbances. This requirement shall apply not only to powered flight, but also to autorotative flight at forward speeds between zero and the maximum forward speed for autorotation. Within the limits of speed specified in 3.2.1 and during the transitions between hovering and the specified extremes, the controls and the helicopter itself shall be free from objectionable shake, vibration, or roughness, as specified in 3.7.1.	Klinar and Craig (Ref. 13) ....Continued $\sigma_u = \sigma_v \doteq 1/5 \ V_w$ (mean) $\sigma_w \doteq 1/15 \ V_w$ (mean) Quickening of display beneficial in <u>still air</u>. No good under gust conditions. AGARD FLT. MECH. PAN. (Ref. 28), AGARD REPT. 408A Specifies that the control required to return the aircraft to trim after a 5 knot or 0.2g acceleration or 5^{o}/sec <u>disturbance</u> should not exceed one-half of the control moment available from trim to stops.
LONGITUDINAL STEADINESS IN HOVER 3.2.2 The helicopter shall be reasonably steady while hovering in still air (winds up to 3 knots), requiring a minimum movement of the cyclic controls to keep the machine over a given spot on the ground, for all terrain clearances up to the disappearance of ground effect. In any case, it shall be possible to accomplish this with less than ±1.0-inch movement of the cyclic controls.	KLINAR and CRAIG (Ref. 13), IAS PAPER 61-60 Correlates pilot rating with stick displacement. The lower the stick displacement, the lower (better) the rating. 

COMMENTS	DISCUSSION AND RECOMMENDATIONS
	Further, results show the hover task to be the most demanding of control power (from trim) even when vertical gusts are considered in forward flight.
Recognizes that the blanket 10 percent margin in 3.2.1 is not realistic; and that margin should be based on (gust) disturbance sensitivity.	Recommendations: The control margin requirements portion of this item should be modified to reflect the disturbance response characteristics of individual helicopters to a design gust (random or deterministic) of specified form and intensity. Research is needed to define design gusts and to establish flight test procedures for demonstrating compliance.
The correlation of pilot rating with pilot's activity in the cited literature does not specify the wind condition, if any, or the speed stability, M_u.	The applicable analytical results are summarized in Paragraphs 2, 3, and 7 in Section V-A. The spec could lead to trouble if it were interpreted to mean that the smaller the control activity in still air, the better the handling qualities. Increasing M_u in still air will greatly reduce control activity, but will strongly increase control activity in gusty air (see Section III-A-3). Tests have shown that pilot rating is much more sensitive to control activity in gusty air. Nevertheless, still air control should provide a good indication of vibration, buffeting, ground effects, etc.; but specifying the stick movements without a corresponding specification on hover position accuracy is too loose. Recommendations: To be a meaningful spec, hovering accuracy (depending on expected use) as well as control displacement should be specified. In addition, item should be expanded to cover hover accuracy and control in a specified gusty environment. Again, the question of design gusts (random in this case) needs research attention.

SPECIFICATION ITEM	APPLICABLE LITERATURE
LONGITUDINAL TRIM EFFECTIVENESS **3.2.3** For all conditions and speeds specified in 3.2.1, it shall be possible in steady-state flight to trim steady, longitudinal control forces to zero. At these trim conditions, the controls shall exhibit positive self-centering characteristics. Stick "jump" when trim is actuated is undesirable.	**CURRY and MATTHEWS (Ref. 29), AHS 20TH ANNUAL NATIONAL FORUM** They recommend that all longitudinal and lateral trim systems be continuous-trim type so that the control forces never fall to zero instantaneously. A hover switch could be used to de-energize trim spring.
LONGITUDINAL FORCE GRADIENT **3.2.4** At all trim conditions and speeds specified in 3.2.1, the longitudinal force gradient for the first inch of travel from trim shall be no less than 0.5 pound per inch and no more than 2.0 pounds per inch. In addition, however, the force produced for a 1-inch travel from trim by the gradient chosen shall not be less than the breakout force (including friction) exhibited in flight. There shall be no undesirable discontinuities in the force gradient, and the slope of the curve of stick force versus displacement shall be positive at all times with the slope for the first inch of travel from trim greater than or equal to the slope for the remaining stick travel.	**CURRY and MATTHEWS (Ref. 29), AHS 20TH ANNUAL NATIONAL FORUM** They recommend that designer select his force gradients on premise that if force applied will indicate the response, then the force should be proportional to steady-state angular velocity and would be similar from one aircraft to another — suggest 2^{o}/sec/lb pitch. New rotorcraft have demonstrated maneuvering stability. Stick force maneuvering stability should be 20 lb/g for typical 3.5g helicopter. **KLINAR and CRAIG (Ref. 13), IAS PAPER 61-60** High stick force gradients were desired for IFR, although pilots were satisfied with zero stick feel for VFR in actual helicopter. **FISHER (Ref. 30), MINISTRY OF AVIATION REPT. AAEE/RES/306** Results for "Whirlwind" show that pilots like 1.5 lb/in. longitudinal and 0.54 lb/in. lateral stick gradients.
ACCELERATION-DECELERATION CHARACTERISTIC **3.2.5** With the helicopter trimmed in steady, level, horizontal flight at maximum forward speed, it shall be possible readily and safely to bring the machine to a quick stop and hover. With the helicopter trimmed in hovering flight, it shall be possible to accelerate rapidly to maximum forward speed, maintaining approximately constant altitude.	

COMMENTS	DISCUSSION AND RECOMMENDATIONS
Cited literature suggests a switch to de-energize trim spring in hover, but means must be provided to maintain servo augmenter backups.	Recommendation: No change; spec seems reasonable.
The consensus of opinion from survey trip was that the gradient forces of Table II were about right, except yaw gradient possibly too low. NATC, Patuxent River, claimed forces too low in all cases. Of course, the survey comments would be dependent on many factors, such as type of mission, stability of long-period mode, speed, etc. During the survey trip, many pilots expressed a desire to add g-force feel to the stick. NATC pilots even found it helpful on low speed approaches.	No experimental support for a universal gradient based on steady rate. The conclusions of Section V-B-2 indicate that pilot desires correlate with angle response in 1 sec rather than with steady rate. Recommendation: No change of longitudinal force or trim requirements. However, consideration should be given to expanding or augmenting this item to include specification requirements for a longitudinal stick acceleration-force feel for helicopters.
	While the spec is not too instructive and there is very little to meet, it serves as a check item. Recommendation: No change.

SPECIFICATION ITEM	APPLICABLE LITERATURE

LONGITUDINAL TRIM CHANGES

3.2.6 Without retrimming, the longitudinal control forces required to change from any trim and power condition to any other trim and power condition as specified in table I, or for performance of the maneuvers discussed in 3.2.5 and 3.5.4 or any other normal helicopter maneuvers, shall not exceed the values given in table II.

CURRY and MATTHEWS (Ref. 29), AHS 20TH ANNUAL NATIONAL FORUM

Recommend the following limit forces as against 8 lb per spec:

	Hover	Transition	Cruise
Long. axis....	10 lb	30 lb	40 lb

Higher for boost, SAS, or "feel" system failures.

BREAKOUT FORCES

3.2.7 With the control trimmed for zero force, breakout forces, including friction in the longitudinal control system, shall conform with the values given in table II when measured in flight.

CURRY and MATTHEWS (Ref. 29), AHS 20TH ANNUAL NATIONAL FORUM

Free play or "slop" should not exceed 1 percent of total travel. Friction and breakout forces should be as shown.

	Target value = minimum	Spec.
Longitudinal.........	0.5 to 2.5	0.5 to 1.5
Collective...........	1.0 to 3.0	1.0 to 3.0
Throttle.............	1.0 to 3.0	—

OBJECTIONABLE CONTROL FORCES

3.2.8 The controls shall be free from objectional transient forces in any direction following rapid longitudinal stick deflections. During and following rapid longitudinal displacement of the control stick from trim, the force acting in a direction to resist the displacement shall not at any time fall to zero. Longitudinal control displacement shall not produce lateral control forces in excess of 20 percent or pedal forces in excess of 75 percent of the associated longitudinal force. For helicopters employing power-boosted or power-operated controls, there shall be no lateral or directional control forces developed.

LONGITUDINAL RESPONSE DELAY

3.2.9 There shall be no objectionable or excessive delay in the development of angular velocity in response to control displacement. The angular acceleration shall be in the proper direction within 0.2 second after longitudinal control displacement. This requirement shall apply for the speed range specified in 3.2.1.

CURRY and MATTHEWS (Ref. 29), AHS 20TH ANNUAL NATIONAL FORUM

The response delay shall not be objectionable if in this time, T_1, a specified threshold value in the proper direction is reached for a step input (achieved in 0.2 sec or less). The threshold may be $\dot{\theta}$ or n_z, depending on speed.

The recommended target value for T_1 is 0.1 sec and a maximum of 0.4 sec to reach either $\dot{\theta} = 0.5^{\circ}/\text{sec}$ or $n_z = 0.01g$.

COMMENTS	DISCUSSION AND RECOMMENDATIONS
Forces seem high for usual one-hand control. No data in Ref. 29 to support figures, although claim "based on recent research flight tests made by industry. the Army and other Government agencies."	Recommendation: No change.
In general agreement with spec; no data to support suggested changes.	Recommendation: No change.
	Recommendation: No change; spec seems reasonable.
The literature appears to be in essential agreement with spec except for maximum value = 0.4 sec (Ref. 29). This seems unreasonably high when it is considered that such delays add directly to pilot's own delay, $\tau \doteq 0.2$ to 0.4 sec.	The high frequency lag of an articulated rotor can be approximated by a pure time delay (good up to 10 rad/sec, as shown by Fig. 31 of Ref. 2) given by $$\tau = \frac{16}{\gamma_1 \Omega}$$ For the average helicopter this time delay is less than 0.1 sec. The product of Lock number, γ_1, and rotor speed tends to remain relatively constant with size; i.e., with increasing size, γ_1 increases while Ω decreases. Thus, the

SPECIFICATION ITEM	APPLICABLE LITERATURE
3.2.9 (continued)	**BREUL (Ref. 27), GRUMMAN REPT. RE-162** For <u>hover</u>, results show control system response time characteristics have a large effect on HQ's in roll and pitch, but none in yaw. Control delays and lags increase damping required most when control sensitivity is high.
LONGITUDINAL STICK STABILITY (POSITION AND FORCE) **3.2.10** The helicopter shall, at all forward speeds and at all trim and power conditions specified in table I, except as noted below, possess positive, static longitudinal control force, and control position stability with respect to speed. This stability shall be apparent in that at constant throttle and collective pitch-control settings a rearward displacement of and pull force on the longitudinal-control stick shall be required to hold a decreased value of steady, forward speed, and a forward displacement and push force be required to hold an increased value of speed. In the speed range between 15 and 50 knots forward, and 10 to 30 knots rearward, the same characteristics are desired, but a moderate degree of instability may be permitted. However, the magnitude of the change in the unstable direction shall not exceed 0.5 inch for stick position or 1.0 pound for stick force. **3.2.10.1** The stability requirements of 3.2.10 are intended to cover all steady flight conditions in which the helicopter might be operated for more than a short time interval. As a guide for the conditions to be investigated, the tabulation of pertinent conditions in table I may be utilized, all referred to the most critical center of gravity position. **3.2.10.2** The helicopter shall not exhibit excessive longitudinal trim changes with variations of rate of climb or descent at constant airspeed. Specifically, when starting from trim, at any combination of power and airspeed within the flight envelope, it shall be possible to maintain longitudinal trim with a longitudinal control displacement of no more than 3 inches from the initial trim position as the engine power or collective pitch, or both, are varied throughout the available range. Generally, the airspeeds needing the most specific investigation of the above characteristics include V_{max} and the speeds between zero and one-half the speed for minimum power.	**CURRY and MATTHEWS (Ref. 29), AHS 20TH ANNUAL NATIONAL FORUM** Recommend control <u>forces</u> should be indicative of the aircraft's forward velocity. Claim rotorcraft have not demonstrated this in the past. They further suggest that in providing static and dynamic stability about all axes, the stick-free or stick force stability is the more important. Stick-fixed or stick position characteristics are important only if relatively large stick motions, either stable or unstable, are required as long as stick force characteristics are stable. **SALMIRS and TAPSCOTT (Ref. 18), NASA TN D-58** Authors conclude that stick force gradients are secondary. **TANAKA and COLVIN (Ref. 31), FTC-TDR-63-32** Authors state in their tests of the HH-43B helicopter that slightly negative or neutral speed stability with respect to control position is not objectionable, provided the control force is positive. **BRAMWELL (Ref. 25), ARC R&M 3104** Points out that the desirability of a positive static margin is not so straightforward for the helicopter as for the fixed-wing. Due to divergent phugoid with the helicopter, the relationships between static and dynamic stability are opposite to fixed-wing aircraft. For fixed-wing aircraft the M_u term is regarded as negligible and static stability depends on M_w. Increasing M_w provides positive static margin and improves dynamic stability. For the helicopter the M_w term is usually small and M_u large; therefore, increasing M_u is responsible for divergent phugoid, so that positive static margin implies dynamic instability.

COMMENTS	DISCUSSION AND RECOMMENDATIONS
Apparently time lags are more critical than time delays (compare Fig. 9 with Fig. 11 of Ref. 27).	(Continued from p. 77) additional time delay caused, for example, by powered control system dynamics must be held to less than about 0.1 sec, which is not unreasonable. Recommendation: No change.
Stick force stability is difficult to analyze for the nonboost helicopter, because the blade forces are hard to predict and can be highly nonlinear.	
Ref. 18 appears to conflict with Refs. 29 and 31 on whether stick force or stick position stability is more important. The discrepant results are most likely due to differences in the task and the inherent stability of the aircraft.	The applicable analytical conclusions are summarised in Paragraphs 19–21 of Section V-A, and are further discussed in Section V-B-6. These are in basic agreement with the suggestions of Refs. 29 and 31 that negative speed position stability is tolerable provided the force speed stability is positive.
Spec item 3.2.10.1 gives emphasis to those flight conditions of 3.2.10 where the helicopter is operated for a long period of time. This, in effect, ties item 3.2.10 to the mission.	Recommendations: Consideration should be given to amending Item 3.2.10 to permit a limited amount of unstable speed position variation. The limiting degree of instability should be the subject of an experimental investigation.
Reference 25 has good summary of the effects of the stability derivatives and coefficients of the stability quartic with respect to speed. However, one must become familiar with the British nondimensional process.	Items 3.2.10.1 and 3.2.10.2 appear reasonable as they apply to item 3.2.10; no change.

SPECIFICATION ITEM	APPLICABLE LITERATURE

LONGITUDINAL DYNAMIC STABILITY

3.2.11 The helicopter shall exhibit satisfactory dynamic stability characteristics following longitudinal disturbances in forward flight. Specifically, the stability characteristics shall be unacceptable if the following are not met for a single disturbance in smooth air:

(a) Any oscillation having a period of less than 5 seconds shall damp to one-half amplitude in not more than 2 cycles, and there shall be no tendency for undamped small amplitude oscillations to persist.

(b) Any oscillation having a period greater than 5 seconds but less than 10 seconds shall be at least lightly damped.

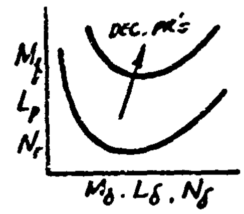
(c) Any oscillation having a period greater than 10 seconds but less than 20 seconds shall not achieve double amplitude in less than 10 seconds.

SALMIRS and TAPSCOTT (Ref. 18), NASA TN D-58

Three-axis test of S-51 (U_0 = 45 knots).

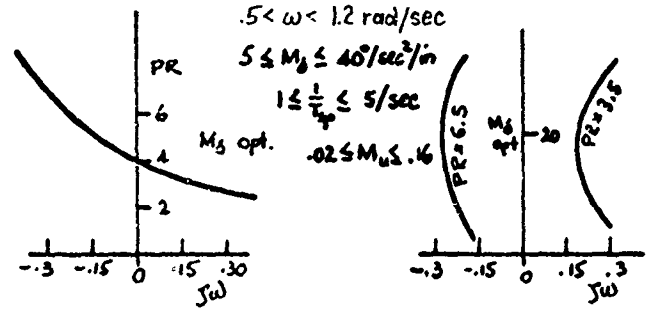
Damping multiplier 1/2, 1, 3, ···, 5 times basic.

Results show handling qualities are improved for increased damping, but are also dependent on control sensitivity.



A'HARRAH and KWIATKOWSKI (Ref. 33), AERO. ENG.

Simulator results near hover correlate pilot rating and control sensitivity with damping factor. For a given control sensitivity, the pilot ratings are shown to be primarily a function of $\zeta\omega$, rather than ζ.

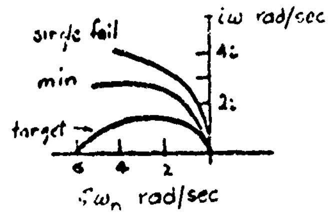


BREUL (Ref. 27), GRUMMAN REPT. RE-162

Presents results from moving-base simulator used to study a tilt wing's handling qualities in still air at hover and during transition (transition results not applicable here). Compares his results on rate damping versus control sensitivity for roll, pitch, and yaw with other investigators and finds that results depend strongly upon the maneuvering task.

CURRY and MATTHEWS (Ref. 29), AHS 20TH ANNUAL NATIONAL FORUM

Show boundaries applicable to longitudinal, lateral, and directional uncoupled motions. Curve based on capabilities and desires of typical pilot.



COMMENTS	DISCUSSION AND RECOMMENDATIONS
Of the literature presented here, only half (Refs. 29, 33) is directly applicable to the question of acceptable cycles or time to damp. The other half is concerned with the pitch damping derivative, M_q, which is specifically covered by item 3.2.14; accordingly the literature and commentary for item 3.2.14 is also "applicable," indirectly, here (see discussion in next column). The boundaries suggested in Ref. 29, which require considerably more damping than the spec, are not supported by any data shown there. The boundaries in AGARD Rept. 408A (Ref. 28), which require increases in ζ over those of the spec of roughly 0.1 and 0.2 for "single-failure" and "normal flight" conditions, respectively, are also unsupported by any presented data. The influence of turbulence on A'Harrah's (Ref. 33) ratings is not clear, although σ_{u_g} = 5 ft/sec was check-tested. At any rate his results are at odds with Seckel's (Ref. 11, item 3.2.14) which are all for σ_{u_g} = 6.3 ft/sec, and large positive M_u. Seckel, <u>et al</u>, conclude that "...it is impossible to express pilot rating as a simple function of $T_{1/2}$...."	The basic problem with this item is that it is a blanket specification which covers conventional phugoid and short-period modes, and also hovering oscillatory modes. The frequency and damping characteristics of these modes vary strongly as a function of M_u and M_q for the usually small values of M_α pertinent to most present-day helicopters (see Table A-II, Appendix A, for approximate relationships). Desirable levels of M_q as a function of M_u (and assuming optimum M_δ) are, however, not consistently expressed in terms of ζ or $\zeta\omega$—i.e., these parameters are <u>not</u>, by themselves, good correlators of handling qualities. The M_δ, M_q, M_u relationship for good hovering control is analyzed in the text and summarized in Section V-B-1, -2. Some of the basic experimental data (Ref. 11) used for the correlations involved show that acceptable ratings (PR = 3.5) were obtained for times to double oscillation amplitudes between about 10 and 25 sec for periods which varied only from about 12 to 14 sec, as $-M_q$ and gM_u increased from 2 to 6 and 0.6 to 1.25. For levels of $-M_q \doteq 1$, considered acceptable for small M_u (Section V-B-1), conventional short-period oscillations, as well as the hovering aperiodic mode, will automatically be reasonably well damped (i.e., $\zeta_{sp}\omega_{sp} \doteq 1$, for $Z_w + M_{\dot\alpha} \doteq -1$; $1/T_{sp_2} \doteq -M_q = 1$). <u>Recommendations:</u> The (a) and (b) portions of this spec should be deleted and covered under item 3.2.14 concerning pitch damping ($-M_q$) requirements. The (c) portion may have validity for unattended operation and should be retained and possibly extended to cover divergent aperiodic modes associated with permissible $-M_u$'s (see 3.2.10 and 3.2.11.2). The entire area needs further experimentation and analysis.

SPECIFICATION ITEM	APPLICABLE LITERATURE

LONGITUDINAL MANEUVER STABILITY

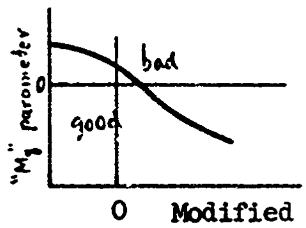
3.2.11.1 The following is intended to insure acceptable maneuver stability characteristics. The normal acceleration stipulations are intended to cover all speeds above that for minimum power required; the angular velocity stipulations shall apply at all forward speeds, including hovering.

(a) After the longitudinal control stick is suddenly displaced rearward from trim a sufficient distance to generate a 0.2 radian/sec. pitching rate within 2 seconds, or a sufficient distance to develop a normal acceleration of 1.5 g within 3 seconds, or 1 inch, whichever is less, and then held fixed, the time-history of normal acceleration shall become concave downward within 2 seconds following the start of the maneuver, and remain concave downward until the attainment of maximum acceleration. Preferably, the time-history of normal acceleration shall be concave downward throughout the period between the start of the maneuver and the attainment of maximum acceleration. Figure 1(a) is illustrative of the normal acceleration response considered acceptable.

(b) During this maneuver, the time-history of angular velocity shall become concave downward within 2.0 seconds following the start of the neuver and the attainment of maximum angular velocity. Figure 1(b) is illustrative of the angular velocity response considered acceptable.

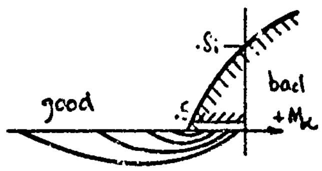
AMER (Ref. 24), NACA REPT. 1200

Results of analysis (peaking of $\dot{n}$ at t = 2.0) are presented in chart form which shows boundary separating combinations of longitudinal stability derivatives that result in satisfactory maneuver stability from combinations that do not. Good prediction for both single-rotor and tandem-rotor in flight test.



SECKEL (Ref. 20), STABILITY AND CONTROL OF AIRPLANES AND HELICOPTERS

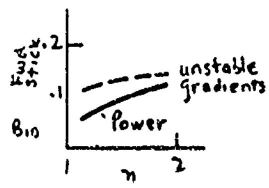
On pp. 363–365 he finds boundary of the "concave downward" requirement ($\ddot{n}$ in t = 2.0) in terms of short-period roots in the s-plane.



Concludes that "concave downward" requirement is a specification on maneuver margin since a <u>reasonable</u> acceleration trim gradient will meet criteria. If trim acceleration gradient (for constant Ω and θ_c under varying Q) is unstable, then maneuver margin is negative and "concave downward" requirement is not satisfied.

$$\frac{B_1}{n}\bigg|_{ss} = \frac{g}{U_0}\,\frac{(U_0M_w - Z_wM_q)}{Z_wM_B - M_wZ_B}$$

Numerator negative for stability

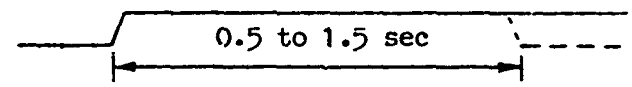
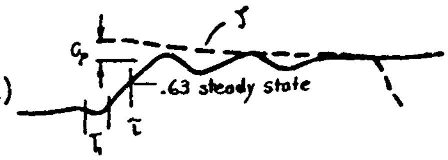
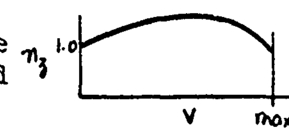
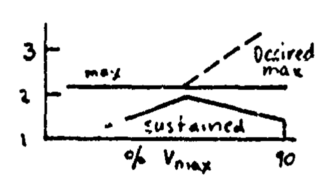
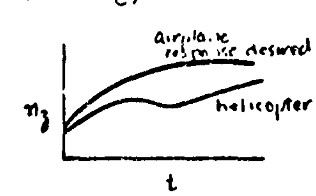


Continued

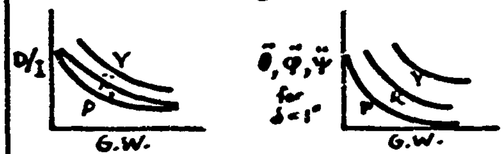
COMMENTS	DISCUSSION AND RECOMMENDATIONS
The analysis of the "concave downward" requirement by Amer is based on satisfying certain functions of the stability derivatives which are rather messy and not very instructive. Although the approach is more informative than the spec, it requires a knowledge of the stability derivatives.	
Seckel's analysis reduces specification to a simple requirement on the maneuver margin, the maneuver margin being a simple expression of the stability derivatives, $M_q Z_w - U_o M_w$.	See page 85.
Continued	

SPECIFICATION ITEM	APPLICABLE LITERATURE
3.2.11.1..................Continued	**TAPSCOTT (Ref. 34), AHS 20TH ANNUAL NAT. FORUM** Criteria were adopted in part because of the difficulty in determining stick position versus acceleration in turn maneuvers. Also, they provide a simple measure of important characteristics found in normal flight — <u>not</u> an oscillatory stability criterion. Claims criterion similar to stability in acceleration flight criteria for the airplane. **BRAMWELL (Ref. 21), RAE REPT. NAVAL 3** Study shows downwash interference of tandem-rotor causes a reversal of stick position with speed, with an associated divergence in dynamic stability. May be eliminated by swashplate dihedral angle at the expense of increasing M_u at hover where downwash effect is absent. States that the tandem can satisfy the NACA "concave downward" requirement, but does not necessarily indicate acceptable maneuverability. Claims the NACA criterion was based on measurements of n_z for a single-rotor helicopter where M_u usually has a small positive value and no purely divergent mode. Suggests that response is determined by dynamic stability and that good response will occur if there is good ζ_{sp} and if the phugoid is no worse than slowly divergent. **BRAMWELL (Ref. 25), ARC R AND M 3104** Develops a maneuver theory analogous to conventional aircraft and claims that the coefficient "C" in the stability quartic gives a good indication of helicopter handling characteristics. The larger the C, the more rapidly the helicopter will reach steady-state. An example (S-51) without tail failed to satisfy the "concave downward" requirement at all speeds for which the maneuver margin is negative and M_u positive. Author states that the maneuvering qualities are vastly improved with a tail which provides negative M_w and increases M_q. Further states that the "E" coefficient may become negative at high U_o, leading to divergence.

COMMENTS	DISCUSSION AND RECOMMENDATIONS
Tapscott says that acceleration trim gradients are hard to determine, whereas Seckel (Ref. 20) claims it is easier than measuring the dynamic response.	
Bramwell is another reference which points out that the "concave downward" requirement does not necessarily indicate acceptable maneuverability. Also infers that the requirement on dynamic stability will determine the response.	The applicable conclusions of this study are summarized in Section V-B-6, which calls attention to the fact that slow aperiodic divergences which are apparently not objectionable (e.g., Ref. 21) are specifically prohibited by this item. Since the intent of this spec is to insure constant-speed maneuvering stability, some modification of the maneuver or the spec language to suppress the effects of speed changes should be considered. For example, the corresponding conventional airplane requirement for stick position and force variations with normal acceleration specifies that the maneuvers be performed at constant speed. <u>Recommendation</u>: Consideration should be given to modifying this specification to eliminate, or recognize the acceptability of, slow divergences due to speed changes.
Bramwell demonstrates that the maneuver margin (coefficient "C") and the "concave downward" requirement yield the same result for the single-rotor helicopter. In regard to his concern for the coefficient "E" becoming negative, a brief survey of single-rotor helicopters shows the sign of "E" is always controlled by M_u for $U_o < 200$ ft/sec. $E \propto M_w Z_u - M_u Z_w$ and $\lvert M_u Z_w \rvert \gg \lvert M_w Z_u \rvert$	

SPECIFICATION ITEM	APPLICABLE LITERATURE
LONGITUDINAL MANEUVER STABILITY **3.2.11.2** To insure that a pilot has reasonable time for corrective action following moderate deviations from trim attitude (as, for example, owing to a gust), the effect of an artificial disturbance shall be determined. When the longitudinal control stick is suddenly displaced rearward from the trim, the distance determined in 3.2.11.1 above, and held for at least 0.5 second, and then returned to and held at the initial trim position, the normal acceleration shall not increase by more than 0.25 g within 10 seconds from the start of the disturbance, except 0.25 g may be exceeded during the period of control application. Further, during the subsequent nosedown motion (with the controls still fixed at trim) any acceleration drop below the trim value shall not exceed 0.25 g within 10 seconds after passing through the initial trim value.	**CURRY and MATTHEWS (Ref. 29), AHS 20TH ANNUAL NATIONAL FORUM** To evaluate transient characteristics, they consider the aircraft short-term response to a step or pulse.  The characteristics of the response which are considered important in achieving good handling qualities for the above input are: $\dot{\theta}$ or n (depending on speed)  T_1 = 0.1 to 0.4 for $\dot{\theta} = 0.5^{\circ}/sec$ or n = 0.01g τ = 0.1 to 1.0 sec; target 0.5 sec O_p = 0.05 ($\dot{\theta}$ or n)$_{steady-state}$ ζ as presented under spec item 3.2.11 by same authors
3.2.12 The response of the helicopter to motion of the longitudinal control shall be such that in the maneuver described in 3.2.11.1, the resulting normal acceleration always increases with time until the maximum acceleration is approached, except that a decrease not perceptible to the pilot may be permitted.	**JENNY (Ref. 35), AHS 20TH ANNUAL NAT. FORUM** Good maneuverability — high speed, low blade loading and power loading. Present spec contains little information on stability and handling qualities based on mission (weapons here) or maneuvers (turns, sustained n_z).  Suggests criteria for maneuverability — ability to pull and sustain a normal load factor. **EDENBOROUGH AND WERNICKE (Ref. 36), AHS 20TH ANNUAL NAT. FORUM** n_z characteristics are found to be most important longitudinal requirements. Two types of n_z — maximum (constant θ_c) quickie; sustained (constant u, Ω, and h, variable θ_c) turns.  

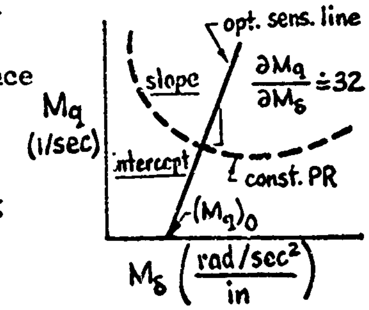
COMMENTS	DISCUSSION AND RECOMMENDATIONS
There are no data in Ref. 29 to support the proposed response characteristics; also, the desired characteristics are not definitely associated with the intent of 3.2.11.2 "...to insure...reasonable time for corrective action...owing to a gust...."	Item 3.2.11.2 seems extraneous in view of other requirements on dynamic and static stability. Furthermore, as stated, it is not indicative of the problem it is supposed to address — the pilot's ability to (comfortably) correct disturbances due to gusts. The closed-loop analyses and correlations discussed in the text are specifically concerned with regulation against gust disturbances — and the resulting requirements on M_q and M_δ should adequately cover this question. On the other hand, the item seems to recognize the possibility of acceptable slowly divergent aperiod or oscillatory modes. Recommendation: Combine with recommended changes to 3.2.11(c).
Reference 35 is concerned with the development of criteria based on weapon systems considerations, which are beyond the scope of this study.	Item 3.2.12 is an additional specification on the shape of the "concave downward" time history (3.2.11.1) designed, apparently, to prevent initial reversals in the time response — seems reasonable. Recommendation: No change.
Reference 36 is concerned with lift and performance rather than stability and control.	

SPECIFICATION ITEM	APPLICABLE LITERATURE
LONGITUDINAL PITCH RESPONSE TO UNIT AND MAXIMUM (CONTROL POWER) CONTROL STEP IN HOVER **3.2.13** Longitudinal control power shall be such that when the helicopter is hovering in still air at the maximum overload gross weight or at the rated power, a rapid 1.0-inch step displacement from trim of the longitudinal control shall produce an angular displacement at the end of 1.0 second which is at least $\frac{45}{\sqrt[3]{W+1000}}$ degrees. When maximum available displacement from trim of the longitudinal control is rapidly applied, the angular displacement at the end of 1.0 second shall be at least $\frac{180}{\sqrt[3]{W+1000}}$ degrees. In both expressions W represents the maximum overload gross weight of the helicopter in pounds.	**TAPSCOTT (Ref. 37), IAS PAPER 60-51** Takes results of NASA TN D-58 and scales them to various types (helicopters and VTOLs) and sizes (weights). Control power and damping given as a function of weight.  **TAPSCOTT (Ref. 34), AHS 20TH ANNUAL NAT. FORUM** Defends the size-dependency $\lvert\sqrt[3]{W+1000}\rvert$ for angular accelerations, since he considers the control response parameter scale in such a manner as to maintain approximately the same linear acceleration at the extremities for a given control input for larger sizes. **EDENBOROUGH and WERNICKE (Ref. 36), AHS 20TH ANNUAL NAT. FORUM** Suggest control effectiveness for weapons helicopters to give $M_\delta/M_q \doteq (\dot\theta/\delta)_{SS} \doteq 14^\circ$ to 20°/sec/in. **LYON (Ref. 38), IAS PAPER 62-63** The mission or type of operation defines control requirements where criteria independent of size and configuration are defined for helicopter-type operation. Develops criterion which agrees with damping vs. control sensitivity plots. Criterion is a pull and recovery maneuver where minimum acceptable value of $\dot\theta/I$ and D/I can be defined by precision maneuver and pilot minimum response time. **KLINAR and CRAIG (Ref. 13), IAS PAPER 61-60** Findings show that dynamic scaling of control sensitivity and damping with size (as indicated in IAS Paper 60-51) did not apply for the large vehicle (35,000 lb). Minimum levels of total control power recommended in the spec were found to be adequate in pitch and yaw, but not for roll, particularly under gusty conditions. However, it was felt that total control power is scalable dynamically with inertia and weight.

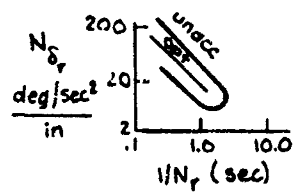
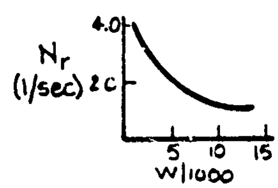
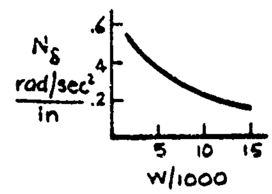
COMMENTS	DISCUSSION AND RECOMMENDATIONS
The variation of requirements with weight suggested in Refs. 37 and 34 is based on specious arguments which are not supported by present data (see Section V-B-2, 3).	
Type of mission appears to be more important than size or weight.	See page 91.
The authors found the minimum requirements as based on weight did not apply to their situation, but failed to realize at this time that the primary factor in determining control sensitivity or power was the speed stability, M_u, in the presence of gust.	

SPECIFICATION ITEM	APPLICABLE LITERATURE
3.2.13 ....Continued	**GARREN (Ref. 39), NASA TN D-2788** VFR investigation of low speed control requirements with Langley variable-stability helicopter showed control power to be the primary factor influencing over-all pilot rating. However, for precision hover task, wide variations in either control power or sensitivity had no appreciable effect on pilot rating. However, these results were obtained in the absence of trim changes and gust effects ($M_u = L_v = M_v = 0$). **DREUL (Ref. 27), GRUMMAN REPT. RE-162** Author concludes that the minimum acceptable control sensitivity (in still air) appears to be independent of task
LONGITUDINAL DAMPING **3.2.14 To insure satisfactory initial response characteristics following a longitudinal control input and to minimize the effects of external disturbances, the helicopter in hovering shall exhibit pitch angular velocity damping (that is, a moment tending to oppose the angular motion and proportional in magnitude to the angular velocity) of at least 8 $(I_y)^{0.7}$ ft-lb/rad/sec, where I_y is the moment of inertia about the pitch axis expressed in slug-ft^2.**	**LOLLAR (Ref. 5), AGARD REPT. 471** Simulator used to measure pilot's transfer function in pitch attitude hover control with $M_u = 0$ (Eq. 9) and gusts injected through M_w: $\frac{\theta}{v_g} = \frac{M_w}{s(s-M_q)}$ $\quad M_w = 0.0175$, $(v_g)_{rms} = 5.1$ ft/sec Results of closed-loop airplane–pilot system has shown a lower bound of airplane damping of 0.75/sec ($1/T_{sp}$) to insure tolerable closed-loop stability for IFR. Optimum gain of approximately 10 in./rad of attitude error. To a first approximation, $\omega_c \doteq 2$ rad/sec over wide range of damping. **SECKEL, ET AL (Ref. 11), PRINCETON REPT. 594** Flight test of variable-stability HUP to determine effect of stability parameters on precision hover tasks in canned turbulence. The effect of M_u was found to be of particular importance in the presence of gust. Control trim gradient and dynamic stability were found to be of secondary import. Their results disclosed greater control sensitivity and higher damping required over other findings (NASA TN D-58)

COMMENTS	DISCUSSION AND RECOMMENDATIONS
The Ref. 39 results are not unexpected when the control power available is limited to levels lower than that desired by the pilot.	On the basis of the applicable results of Paragraphs 5, 8, and 9, Section V-A and V-B-2 and -3 the parameter involved in this criterion (response in 1 sec) is a good one, but its value as a function of mission is inadequately expressed by the $\sqrt[3]{W+1000}$ factor; also, no consideration is given to the effects of gust sensitivity (M_u) which strongly influence the desired M_δ (but maybe not the desired 1 sec response — see discussion item 3.3.5). Recommendations: Consideration should be given to rewriting this item to more adequately reflect mission and gust sensitivity effects on required response. Continued experiments and analyses are required.
Breul's optimum control sensitivity line on plot of M_q versus M_δ gives constant $M_\delta \doteq 0.2$ to 0.3 rad/sec^2/in. (infinite slope). Value consistent with, but slope at odds with, other investigations.	
The single-loop attitude control problem of Ref. 5 may not be truly representative of the pitch control in hover for $M_u \neq 0$. However, the results show the same trends as do the more complete analyses of Section III.	See page 93.
The test of Ref. 11 was one of the first to give pilot ratings for the closed-loop hover task, and to show the importance of M_u in the presence of gust.	

SPECIFICATION ITEM	APPLICABLE LITERATURE
3.2.14 .... Continued	**MILLER and CLARK (Ref. 14), AIAA PAPER 64-618** Fixed-base UAC simulator study reveals factors which define the slope and intercept of optimum sensitivity control line on M_q versus M_δ pilot rating contours for hover. For <u>compensatory hover</u> with gust inputs, the slope (approximately 32) is independent of gust level and the intercept is mostly a function of M_u and gust level and, to a lesser extent, X_u. (Roll dynamics fixed in good region with low L_v.) Concluded that the oscillatory mode dynamics (ζ_p, ω_p) are of minor importance within the satisfactory range of handling qualities for both compensatory and pursuit tracking tasks. 
3.3 DIRECTIONAL-LATERAL **DIRECTIONAL CONTROL ON THE GROUND** **3.3.1** Directional control shall be sufficiently powerful, in order that its use in conjunction with the other normal controls will permit easy execution of all normal taxiing maneuvers with wheel gear on land and float gear in water using normal rotor speeds. In particular, the following ground handling conditions shall be met: (a) It shall be possible, without the use of brakes, to maintain a straight path in any direction in a wind of 35 knots. (b) It shall be possible to make a complete turn in either direction by pivoting on one wheel in a wind of 35 knots. **TRANSLATIONAL VELOCITY OF 35 KNOTS** **3.3.2** From the hovering condition, it shall be possible to obtain steady, level, translational flight at a sidewise velocity of 35 knots to both the right and the left. At the specified sidewise velocity and during the transition from hovering, the controls and the helicopter itself shall be free from objectionable shake, vibration, or roughness as specified in 3.7.1	

COMMENTS	DISCUSSION AND RECOMMENDATIONS
This work is a further verification of the results Seckel obtained above, only on a fixed-base simulator. More detailed information is given on the effects of M_u and X_u in the presence of gust.	The results of this study, summarized in Paragraphs 1 and 6 of Section V-A and in V-B-1, indicate that a minimum pitch damping of about $-M_q \doteq 1$ is required independent of mission or size; and values greater than this may be necessary, depending on the values of M_u and M_δ. For vehicles not subject to gust upsets (of academic interest to a helicopter spec), M_q may be allowed to approach zero. <u>Recommendation</u>: This item should be modified to reflect a minimum requirement, regardless of size or inertia, of $M_q \doteq -1/\text{sec}$.
No comments were recorded concerning the contents of this spec on survey trip.	<u>Recommendation</u>: No change; spec item seems reasonable.
No comments were recorded concerning the contents of this spec on survey trip.	<u>Recommendation</u>: No change; spec item seems reasonable.

SPECIFICATION ITEM	APPLICABLE LITERATURE
DIRECTIONAL STEADINESS IN HOVER 3.3.3 The requirements of 3.2.2 shall be applicable to lateral as well as to longitudinal control motions. It shall be possible to meet this requirement with less than ±1-inch movement of the directional control.	See literature under items 3.2.2, 3.3.5, and 3.3.6.
LATERAL CONTROL POWER (CONTROL MOMENT AVAILABLE) 3.3.4 In all normal service loading conditions, including those resulting in asymmetrical lateral center of gravity locations and steady flight under the conditions specified in 3.2.1 (including autorotation) and 3.3.2, a sufficient margin of control effectiveness, and at least adequate control to produce 10 percent of the maximum attainable hovering rolling moment shall remain at each end.	See literature under item 3.2.1
DIRECTIONAL YAW RESPONSE TO UNIT AND MAXIMUM (CONTROL POWER) CONTROL STEP 3.3.5 Directional control power shall be such that when the helicopter is hovering in still air at the maximum overload gross weight or at rated takeoff power, a rapid 1.0-inch step displacement from trim of the directional control shall produce a yaw displacement at the end of 1.0 second which is at least $\frac{110}{\sqrt[3]{W+1000}}$ degrees. When maximum available displacement from trim of the directional control is rapidly applied at the conditions specified above, the yaw angular displacement at the end of 1.0 second shall be at least $\frac{330}{\sqrt[3]{W+1000}}$ degrees. In both equations W represents the maximum overload gross weight of the helicopter in pounds.	**A'HARRAH and KWIATKOWSKI (Ref. 33), AERO. ENG.** Has single-loop simulator results of N_r versus N_{δ_r} for single degree of freedom in hover.  **TAPSCOTT (Ref. 37), IAS PAPER 60-51** Scales N_r versus N_{δ_r} results of NASA TN D-58 to any size.   **DAW (Ref. 16), MRCC REPT. LR-400** See text, Sections III and IV.

COMMENTS	DISCUSSION AND RECOMMENDATIONS
See comments under items 3.2.2, 3.3.5, and 3.3.6.	The applicable conclusions of this study are summarized in Paragraphs 10, 2, 3, and 7, Section V-A. Discussion and <u>Recommendations</u> for item 3.2.2 apply here as well.
See comments under item 3.2.1	Discussion and <u>Recommendations</u> for item 3.2.1 apply here as well.
The ratings of this reference were taken at hover in still air as spec suggests. However, the ratings are independent of weight and inertia just as any handling qualities test conducted in a simulator must be.	The applicable analytical results are summarized in Paragraphs 10, 13, 14, and 15 of Section V-A, and in V-B-2 and -3 (also see item 3.2.13). The interesting thing about the V-B-2(c) conclusion is its implication that the optimum response in 1 sec per inch of rudder is largely unaffected by gust sensitivity, so that this metric apparently has some <u>general</u> validity, independent of the derivatives involved. Discussion given for 3.2.13 is also applicable here. <u>Recommendations</u>: Same as for 3.2.13.
The results were not obtained at hover as the spec requires, and the dynamics of the remaining axes (longitudinal and lateral) were not optimally set. The latter fact could have a significant influence on the pilot's rating about the directional axis.	
Fully discussed in Sections III and IV.	

SPECIFICATION ITEM	APPLICABLE LITERATURE

DIRECTIONAL CONTROL MARGIN FOR 35 KNOT WIND AT CRITICAL DIRECTION

3.3.6 It shall be possible to execute a complete turn in each direction while hovering over a given spot at the maximum overload gross weight or at takeoff power (in and out of ground effect), in a wind of at least 35 knots. To insure adequate margin of control during these maneuvers, sufficient control shall remain at the most criti il azimuth angle relative to the wind, in order that, when starting at zero yawing velocity at this angle, the rapid application of full directional control in the critical direction results in a corresponding yaw displacement of at least $\frac{119}{\sqrt[3]{W+1000}}$ degrees in the first second, where W represents the maximum overload gross weight of the helicopter in pounds.

DAW and McGREGOR (Ref. 16), NRCC REPT. LR-400

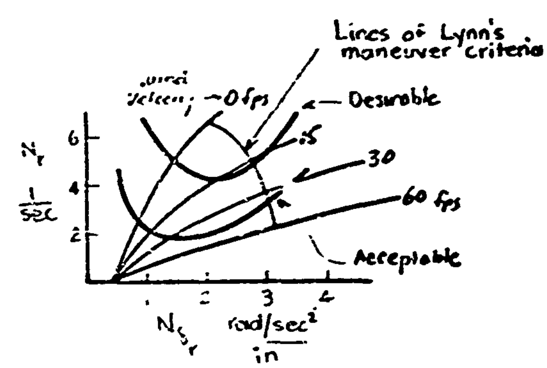
The amount of rudder necessary to hover crosswind is a direction function of N_v and N_{δ_r},

$$\delta_{r_o} = \frac{V_o N_v}{N_{\delta_r}}$$

Roughly speaking, configurations requiring full rudder to hover in a 15 knot crosswind were rated 6.5; those requiring 1 inch of rudder pedal were eligible for a 3.5 rating providing dynamics were acceptable.

LYNN (Ref. 38), IAS PAPER 62-63

Concludes that definition of yaw control criteria and the interpretation of related tests must involve consideration of the gust sensitivity and the operational wind conditions.



For the example shown in the figure, with winds above 15 ft/sec, the maneuver criteria fall outside the desirable region which would be reflected in the pilot's rating.

COMMENTS	DISCUSSION AND RECOMMENDATIONS
	This item combines requirements for trim and maneuverability in a fairly logical way, and the form of this specification is basically good. However, the required maneuverability as a function of mission is probably inadequately specified by the chosen numerical function of weight. Recommendations: Same as 3.2.13, i.e., the criterion values should be specified in terms of mission and/or gust sensitivity rather than weight.
The consensus of opinion from the survey trip was that the 35 knot sidewind set the maximum control power in yaw. The criteria as proposed by Lynn for yaw control sensitivity agrees with the result found by Daw in the preceding literature, i.e., the control sensitivity must not only combine with the proper damping, but must consider the speed stability, N_V (gust sensitivity), and available control in the presence of wind.	

SPECIFICATION ITEM	APPLICABLE LITERATURE
MAXIMUM DIRECTIONAL SENSITIVITY IN HOVER, LIGHTEST NORMAL SERVICE LOADING **3.3.7** The response of the helicopter to directional-control deflection, as indicated by the maximum rate of yaw per inch of sudden pedal displacement from trim while hovering shall not be so high as to cause a tendency for the pilot to overcontrol unintentionally. In any case, the sensitivity shall be considered excessive if the yaw displacement is greater than 50 degrees in the first second following a sudden pedal displacement of 1 inch from trim while hovering at the lightest normal service loading.	**EDENBOROUGH and WERNICKE (Ref. 36), AHS 20TH ANNUAL NAT. FORUM** For precision yaw control (weapons firing) a short response time is needed, which implies a rate type of control. Points out that helicopters, especially during hover, approach a pure acceleration control. Suggest control sensitivities of the order of $30^{\circ}-50^{\circ}$/sec/in. and response times $-1/N_r < 0.2$ sec for hover, depending on weapons system.
COORDINATED TURNS IN AUTOROTATION **3.3.8** It shall be possible to make coordinated turns in each direction while in autorotation, at all autorotation speeds.	
LATERAL STABILITY — N_{β}, L_{β}, δ_r, δ_a VERSUS β LINEAR; 10 PERCENT MARGIN IN LATERAL CONTROL **3.3.9** The helicopter shall possess positive, control fixed, directional stability, and effective dihedral in both powered and autorotative flight at all forward speeds above 50 knots, 0.5 V_{max}, or the speed for maximum rate of climb, whichever is the lowest. At these flight conditions with zero yawing and rolling velocity, the variations of pedal displacement and lateral control displacement with steady sideslip angle shall be stable (left pedal and right stick displacement for right sideslip) up to full pedal displacement in both directions, but not necessarily beyond a sideslip angle of 15 degrees at V_{max}, 45 degrees at the low speed determined above, or beyond a sideslip angle determined by a linear variation with speed between these two angles. Between sideslip angles of ±15 degrees, the curve of pedal displacement and lateral control displacement plotted against sideslip angle shall be approximately linear. In all flight conditions specified above, a 10 percent margin of both lateral and longitudinal control effectiveness (as defined in 3.2.1 and 3.3.4) shall remain.	

COMMENTS	DISCUSSION AND RECOMMENDATIONS
Single rotors, due to higher damping, do appear to be a rate control at hover, but with a time constant of 3 to 4 sec. Tandem rotors are more of an acceleration type of control with approximately zero damping in yaw. These are typical of basic values without augmentation	Recommendation: No change; the spec appears reasonable.
	Recommendation: No change; reasonable requirement.
	Recommendation: No change; the present requirements for linearity, sign, etc., are appropriate.

SPECIFICATION ITEM	APPLICABLE LITERATURE

ROLLING REVERSAL OF $p \rightarrow \delta_a$

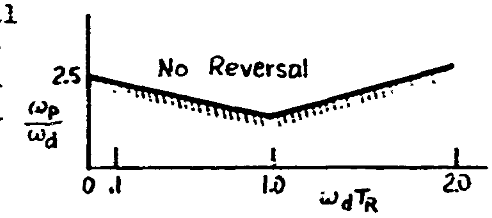
3.3.9.1 At the conditions specified in 3.3.9, it shall be possible to make complete turns in each direction with pedals fixed, by use of cyclic control stick alone. At all speeds specified in 3.3.9, no reversal of rolling velocity (i.e., return through zero) shall occur after a small lateral step displacement of the control stick is made with pedals fixed. The stick deflection chosen shall be such that the maximum angle of bank reached during 6 seconds is approximately 30 degrees. This requirement is intended to apply to angular velocity type controls.

ADVERSE YAW

3.3.9.2 During pedal fixed rolling maneuvers, there shall be no objectionable adverse yaw.

ASHKENAS and McRUER (Ref. 4), WADC-TR-59-135

The amount of Dutch roll appearing in the rolling motion following a step aileron depends primarily on departure of $(\omega_\phi/\omega_d)^2$ from unity. Shows figure for incipient "p" reversal, assuming long spiral mode time constant, T_s, and low dutch roll damping, ζ_d.

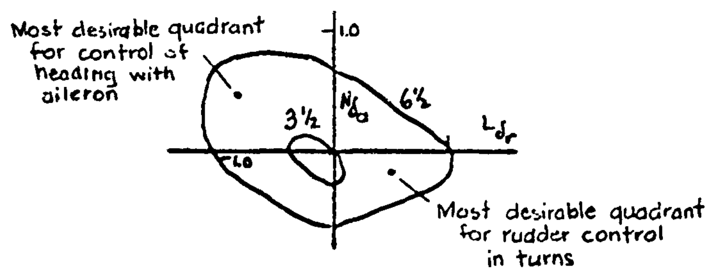


CROME and A'HARRAH (Ref. 51), IAS AERO. ENG., SEPT. 1960

Among other things, they show that acceptable conventional aircraft pilot ratings are obtained (in a fixed-base simulator) only for $|p_1|/|p_{ss}| < 0.045$; where p_1 is the amplitude of roll oscillation at the first overshoot following a step aileron input.

McGREGOR (Ref. 42), NRCC REPT. LR-390

Results from airborne simulator show good handling qualities region confined to the left plane where right δ_r produces left rolling moment. L_r negative and N_r positive.



Pedal-fixed considerations are given by the handling qualities along the N_{δ_a} axis ($L_{\delta_r} = 0$).

Note: N_{δ_r} is positive for right rudder and L_{δ_a} is positive for right aileron.

GARREN and KELLY (Ref. 15), NASA TN D-2477

DAW (Ref. 16), NRCC REPT. LR-400

See Section III.

COMMENTS

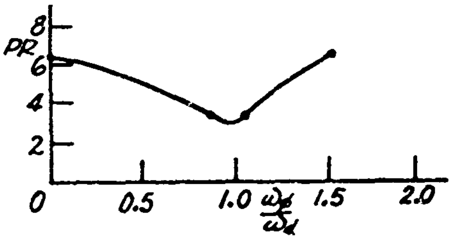
The references cited here, except for the last two (Refs. 15, 16) apply to both requirements.

Reference 4 relates incipient roll reversal (for rate control situations) to the aileron yaw characteristics through

$$(\omega_\varphi/\omega_d)^2 \doteq 1 - (N_{\delta_a}/L_{\delta_a})(L_v/N_v)$$

For $(\omega_\varphi/\omega_d)^2 < 0.5$, implying negative $N_{\delta_a}/L_{\delta_a}$ (adverse yaw) for normally negative L_v/N_v, reversals will occur for small dutch roll damping.

References 51 and 42 indicate that the limit of acceptability occurs before actual reversal and is a function of the relative magnitude of the roll oscillation which is related to departures of ω_φ/ω_d from 1.0. For example, the Ref. 42 results can be converted (for $L_{\delta_r} = 0$) into the following trends:



These trends are consistent with those for conventional aircraft collected in Ref. 3.

Finally, Refs. 15 and 16, as analyzed in Section III-B, indicate that considerations other than adverse aileron yaw are important in achieving good heading control during approach.

DISCUSSION AND RECOMMENDATIONS

The applicable analytical results are summarized in Paragraph 22 of Section V-A and in V-B-5.

A limited survey shows the single rotor generally has roll rate control, whereas the tandem configuration has roll position control with lateral stick inputs commanding bank angle and associated turn rate instead of roll rate. The latter characteristics involve roll rate reversals which are acceptable provided bank angle oscillations are small and initial response is reasonably fast (see Fig. 25a, b, c).

If the spec is intended to apply only to roll rate control situations, as implied by the last sentence, it seems reasonable, but incomplete in view of unacceptable roll rate oscillations of either sign shown in the literature.

Recommendations: Consideration should be given to revising the language to clarify intended area of application (i.e., rate control) and expanding to cover generally undesirable roll rate oscillations; also expanding to cover bank-angle-commanded (rather than roll rate) dynamic situations.

The spec is OK as a check item, but is non-instructive and indefinite.

Recommendations: Spec should be rewritten to specify "objectionable" yaw characteristics, e.g., in terms of allowable sideslip or allowable time delay in achieving heading rate in correct direction (see Section V-B-4).

SPECIFICATION ITEM	APPLICABLE LITERATURE
LATERAL AND DIRECTIONAL TRIM FORCES **3.3.10** For all conditions and speeds specified in 3.2.1 and 3.3.2, it shall be possible in steady flight to trim steady lateral and directional control forces to zero. At these trim conditions, the controls shall exhibit positive self-centering characteristics. Stick "jump" when trim control is actuated is undesirable.	
LATERAL AND DIRECTIONAL TRIM GRADIENTS **3.3.11** At all trim conditions and speeds specified in 3.3.10, the lateral force gradient for the first inch of travel from trim shall be no less than 0.5 pound per inch and no more than 2.0 pounds per inch. In addition, however, the force produced for a 1-inch travel from trim by the gradient chosen shall not be less than the breakout force (including friction) exhibited in flight. The slope of the curve of stick force versus displacement shall be positive at all times and the slope for the first inch of travel from trim shall always be greater than or equal to the slope for the remaining stick travel. The directional control shall have a limit force of 15 pounds at maximum deflection with a linear force gradient from trim position. There shall be no undesirable discontinuities in either the lateral or directional force gradients.	**CURRY and MATTHEWS (Ref. 29), AHS 20TH ANNUAL NATIONAL FORUM** They recommend higher limit forces for the directional control than the spec. (see table below)
LATERAL AND DIRECTIONAL FORCE CHANGES **3.3.12** From trimmed initial conditions, the lateral and directional control forces required for the performance of the maneuvers discussed in 3.2.6, 3.3.1, 3.3.2, 3.3.4, 3.3.5, 3.3.6, 3.3.8, and 3.3.9.1, shall conform with the values given in table II.	**AGARD REPT. 408 (Ref. 28)** Recommends the following breakout forces: (see table below)

Curry and Matthews (Ref. 29):

	Hover	Transition	Cruise
Lateral........	7 lb	15 lb	20 lb
Directional....	30 lb	60 lb	100 lb

AGARD Rept. 408 (Ref. 28):

	Normal	Boost Failure
Lateral...............	0.5 – 2.0	< 4
Directional..........	1.0 – 10	< 15

COMMENTS	DISCUSSION AND RECOMMENDATIONS
	Recommendation: No change; spec is reasonable.
The literature suggests higher limit forces for directional control than those of spec. This suggestion coincides with comments obtained on the survey trip.	Recommendation: Amend requirement to permit larger directional control forces.
	As noted above, the directional control force limit in Table II (15 lb) appears too low. Recommendation: Amend Table II to show larger permissible directional control forces.

SPECIFICATION ITEM	APPLICABLE LITERATURE

LATERAL AND DIRECTIONAL BREAKOUT FORCES

3.3.13 With the controls trimmed for zero force, the breakout forces including friction in the lateral and directional control systems shall conform with the values given in table II when measured in flight.

AGARD REPT. 408 (Ref. 28)

CURRY and MATTHEWS (Ref. 29), AHS 20TH ANNUAL NAT. FORUM

Both recommend the following breakout forces:

	Normal	Boost Failure
Lateral..........	0.5–2.0	<4
Directional......	1.0–10	<15

CONTROL FORCE CROSS-COUPLING

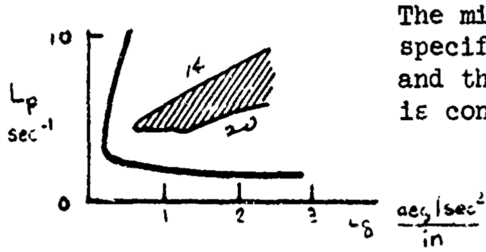
3.3.14 The controls shall be free from objectionable transient forces in any direction following rapid lateral stick or pedal deflections. During and following a rapid lateral displacement of the control stick from trim or a rapid pedal displacement from trim, the force acting in a direction to resist the displacement shall not at any time fall to zero. Lateral control displacement shall not produce longitudinal control forces in excess of 40 percent or pedal forces in excess of 100 percent of the associated lateral force. Pedal displacement shall not produce longitudinal control forces in excess of 8 percent or lateral control forces in excess of 6 percent of the associated pedal force. For helicopters employing power-boosted or power-operated controls, there shall be no longitudinal control forces developed in conjunction with lateral or directional control displacement.

MAXIMUM P/Inch δ_a

3.3.15 The response of the helicopter to lateral-control deflection, as indicated by the maximum rate of roll per inch of sudden control deflection from the trim setting, shall not be so high as to cause a tendency for the pilot to overcontrol unintentionally. In any case, at all level flight speeds, including hovering, the control effectiveness shall be considered excessive if the maximum rate of roll per inch of stick displacement is greater than 20 degrees per second.

EDENBOROUGH and WERNICKE (Ref. 36), AHS 20TH ANNUAL NAT. FORUM

Claim the roll response requirement established for an armed helicopter is 14°–20°/sec/in.



The minimum was not specified in the spec and the maximum limit is considered because of L_p and its effect on ease of trim during steady flight.

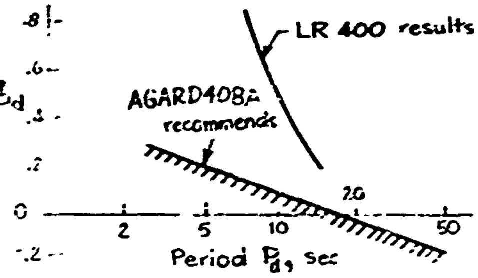
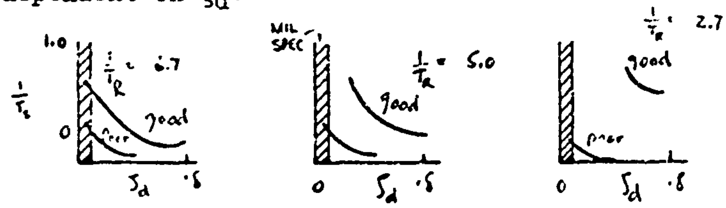
3.3.16 There shall be no objectionable or excessive delay in the development of angular velocity in response to lateral or directional control displacement. The angular acceleration shall be in the proper direction within 0.2 second after control displacement. This requirement shall apply for all flight conditions specified in 3.2.1, including vertical autorotation.

See literature under item 3.2.9.

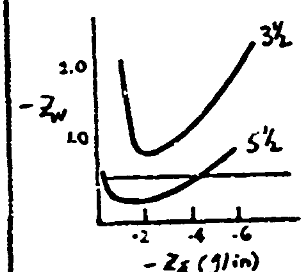
COMMENTS	DISCUSSION AND RECOMMENDATIONS
The values recommended in the literature extend slightly the "normal" values given in Table II.	Recommendation: No change warranted on basis of available data; spec appears reasonable.
	Recommendation: No change; spec appears reasonable.
Reference 36 agrees with maximum roll rate of spec, but also suggests a minimum maneuvering requirement for an armed helicopter. However, maximum limit appears to involve low L_p considerations which are not appropriate to this spec item.	Excessive control sensitivity will always lead to degraded pilot ratings and some limit is desirable; however this spec limits roll rates to values considerably below those for conventional fighter aircraft — i.e., for 5 inches of full stick, 3.3.15 results in $p_{max} = 100^{\circ}/sec$, whereas Ref. 51 requires 100° in 1 sec and considers the limit to be $220^{\circ}/sec$. While such rates appear excessive for helicopters, they can apparently be accommodated without overcontrol tendencies. Recommendation: Amendment of the limiting value, especially for high speed conditions, should be considered.
See comments under item 3.2.9.	See discussion of item 3.2.9. Recommendation: No change.

SPECIFICATION ITEM	APPLICABLE LITERATURE
LATERAL TRIM CHANGES DUE TO POWER AND COLLECTIVE PITCH **3.3.17** The helicopter shall not exhibit excessive lateral trim changes with changes in power or collective pitch, or both. Specifically, when starting from trim at any combination of power and airspeed within the flight envelope of the helicopter, it shall be possible to maintain lateral trim with a control displacement amounting to no more than 2 inches from the initial trim position as the engine power or collective pitch, or both, are varied either slowly or rapidly in either direction throughout the available range.	**TAPSCOTT (Ref. 34), AHS 20TH ANNUAL NAT. FORUM** Analysis of a large helicopter (30,000 lb) showed that the factor completely overriding any maneuver consideration was the need to cancel yaw moments brought about by power changes.
LATERAL ROLL RESPONSE TO UNIT AND MAXIMUM (CONTROL POWER) CONTROL STEP **3.3.18** Lateral control power shall be such that when the helicopter is hovering in still air at the maximum overload gross weight or at the rated power, a rapid 1-inch step displacement from trim of the lateral control shall produce an angular displacement at the end of one-half second of at least $\frac{27}{\sqrt[3]{W+1000}}$ degrees. When maximum available displacement from trim of the lateral control is rapidly applied at the conditions specified above, the resulting angular displacement at the end of one-half second shall be at least $\frac{81}{\sqrt[3]{W+1000}}$ degrees. In both expressions W represents the maximum overload gross weight of the helicopter in pounds.	See literature under item 3.2.13. **ASHKENAS (Ref. 3), AGARD REPT. 533** Shows that best opinion in-flight roll power for helicopters/VTOLs corresponds to $\varphi_1 \doteq 0.75$ rad. For tests where maximum deflection was not used, converting to φ_1 per inch (assuming 5 inches of cockpit travel) gives 0.15 rad $\doteq 10^{\circ}$ (see Section IV).
LATERAL AND DIRECTIONAL DAMPING **3.3.19** To insure satisfactory initial response characteristics following either a lateral or directional control input and to minimize the effect of external disturbances, the helicopter, in hovering, shall exhibit roll angular velocity damping (that is, a moment tending to oppose the angular motion and proportional in magnitude to the rolling angular velocity) of at least $18(I_x)^{0.7}$ ft-lb/rad/sec., where I_x is	See applicable literature under items 3.2.14 and 3.3.5. **ASHKENAS (Ref. 3), AGARD REPT. 533** Correlates desirable levels of roll and yaw damping (L_p, N_r) for single-degree-of-freedom situations (as in hover) and finds acceptable levels to be near −1.0 (same as for M_q)

COMMENTS	DISCUSSION AND RECOMMENDATIONS
It would seem, from the Ref. 34 comment, that similar limits should be placed on the directional controls.	Recommendation: Requirements, similar in wording, etc., to present spec, should be added to cover pedal control limits.
See Section IV.	The applicable conclusions of this study are summarized in Paragraphs 10, 5, 8, and 9 of Section V-A and in V-B-2 and -3. The use of a 1/2 sec, rather than a 1 sec, response criterion, places a premium on aileron deflection rate, is not consistent with the correlations of Ref. 3 presented in Section IV, and represents a difficult flight test procedure. The discussions and Recommendations of item 3.2.13 apply here as well.
See applicable comments under items 3.2.14 and 3.3.5.	See applicable results under items 3.2.14 and 3.3.5.

SPECIFICATION ITEM	APPLICABLE LITERATURE
3.3.19....................Continued	**GARREN and KELLY (Ref. 15), NASA TN D-2477**
the moment of inertia about the roll axis expressed in slug-ft². The yaw angular velocity damping should preferably be at least $27(I_z)^{0.7}$ ft-lb/rad/sec., where I_z is the moment of inertia about the yaw axis expressed in slug-ft².	Results, in addition to those given in the text, text, are as follows: N_β should be > 0.3. Minimum satisfactory N_{δ_r} and N_r are in agreement with current criteria ($N_{\delta_r} \doteq 0.2$ and $N_r \doteq -1$). For optimum characteristics at some $N_\beta > 0.3$, add sufficient N_r to yield $0.8 < \zeta^* < 0.1$. (ζ^* defined as $N_r/2\sqrt{N_\beta}$.) For IFR, reduced L_v results in improved handling qualities due to less disturbances and/or oscillations; for VFR, pilot was insensitive to L_v effect. **DAW and McGREGOR (Ref. 16), NRCC REPT. LR-400** In addition to results discussed in text, they show that ζ_d versus dutch roll period, P_d (see below), tends to be meaningless, since the pilot's main objection as damping is decreased is based on the disturbances and not on the periodic nature of the motion.  **GONZALES (Ref. 40), PRINCETON UNIV. REPT. 457** **GOLDBERG (Ref. 41), PRINCETON UNIV. REPT. 496** **PRINCETON VARIABLE-STABILITY HELICOPTER (HUP-1) IN SMOOTH AIR** For good handling qualities, ω_d should be well damped. $1/T_s$ may be slightly unstable, but high positive stability desired if Dutch roll damping is low. Therefore, desirable spiral characteristics are dependent on ζ_d. 

COMMENTS	DISCUSSION AND RECOMMENDATIONS
The desirable levels of yaw and roll damping (the subject of this spec item) are masked in the results cited here which are expressed in terms of dutch roll frequency and damping. Dutch roll damping by itself has not been found to be a good metric of handling qualities in the listed literature. The use of ζ^* (Ref. 15) is limited, for it does not indicate poor results when N_β becomes small due t loss of the $\dot{\psi} \rightarrow \delta_a$ effectiveness in turns (see Section III for other correlations of these data). The requirement for minimum N_β (from Ref. 15) may not be associated with dutch roll frequency and damping, but rather with turn coordination questions (Fig. 14 of text). Spiral characteristics are of course influenced by N_β and N_r; the desirable properties shown in Refs. 40 and 41 may be due to good roll angle response characteristics as discussed in Section IV and 3.3.9.1.	The basic rotary damping requirements do not appear to be significantly size-, mission-, or weight-dependent. Basic requirements for minimum L_p and N_r appear to be about −1, but these may be low for high gust sensitivity and correspondingly high control effectiveness. The spec currently does not address questions of adequate damping and frequency for oscillatory lateral modes, although the literature has a number of attempted correlations (unsuccessful in general) based on these or related parameters. Also, AGARD Rept. 408 recommends the same damping–frequency relationships for lateral as for longitudinal oscillations. All this is very reminiscent of the discussions concerning items 3.2.11 and 3.2.14 which apply equally well here, with suitable changes in axes. Recommendations: This item should be modified to reflect minimum requirements, regardless of size or inertia of $L_p = N_r \doteq -1$/sec.

SPECIFICATION ITEM	APPLICABLE LITERATURE
MINIMUM DUTCH ROLL FREQUENCY	See 3.3.19.
VERTICAL STEADINESS **3.4.1 It shall be possible to maintain positive control of altitude within ±1.0 foot by use of the collective-pitch control while hovering at constant rotor rpm under the conditions of 3.2.2. This shall be accomplished with a minimum amount of collective stick motion required, and in any case it shall be possible to accomplish this with less than ±¼ inch movement of the collective stick. When a governor is employed, there shall be no objectional vertical oscillation resulting from lag in governor response.**	**GARREN (Ref. 43), NASA TN D-1488** Shows iso-opinion plots for Z_w vs. Z_δ. Optimum value between 0.25g/in. for Z_w = 1 to 0.15g/in. for Z_w = 0.  If Z_w is high, good handling qualities for $T/W \geq 1.08$; if T/W is high, good handling qualities for $-Z_w \geq 0.2$. Increasing time delay results in deterioration of controllability. Regardless of damping, $\tau > 0.4$ gave poor control. **A'HARRAH and KWIATKOWSKI (Ref. 33), AERO. ENG.** Results show optimum gains, Z_δ, less than 0.3g/in. for damping less than 1 ($-Z_w$ = 1 sec^{-1}).
COLLECTIVE PITCH CONTROL 3.4.2 The collective-pitch control shall remain fixed at all times unless moved by the pilot and shall not tend to creep, whether or not cyclic or directional controls are moved. The maximum effort required for the collective control shall not exceed the values specified in table II. The breakout force (including friction) shall be within the acceptable limits as specified in table II.	
COLLECTIVE TO CYCLIC CONTROL FORCE COUPLING 3.4.3 Movement of the collective-pitch control shall not produce objectionable forces in the cyclic control; in no case shall these forces exceed 1 pound. In helicopters where power-operated or power-boosted controls are utilized, there shall be no control force coupling.	

COMMENTS	DISCUSSION AND RECOMMENDATIONS
See 3.3.19.	There is no lateral-directional spec equivalent to the item 3.2.11.1 longitudinal maneuvering criterion, so that the question of minimum "stiffness" or frequency is not treated in the current spec. <u>Recommendation</u>: Research and analysis be conducted to define the minimum acceptable dutch roll frequency.
Both references tend to show optimum control sensitivity, Z_δ, at about 0.25g/in. for the typical rotor damping of $-Z_w \doteq 0.5\ \mathrm{sec}^{-1}$. However, the optimum damping appears to be greater than 1 sec^{-1} ($-Z_w > 1$) in hover, consistent with correlations in Ref. 3. The optimum control line in this case also indicates a constant altitude in 1 sec, h_1.	The closed-loop situation is similar to the single-degree-of-freedom hover mode in pitch, roll, and yaw; but values of $-Z_w < 1$, while not "optimum," appear acceptable (PR < 3.5) <u>Recommendation</u>: No change; spec is reasonable.
	<u>Recommendation</u>: No change; spec appears reasonable.
	<u>Recommendation</u>: No change; spec appears reasonable.

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APPENDIX A

APPROXIMATE FACTORS

The formulation of approximate factors for helicopters is more complicated than for conventional aircraft. The extra complexity stems from the addition of low speed flight. In this flight regime certain important stability derivatives differ greatly from their values at conventional aircraft speeds and also differ substantially from one type of helicopter to another. As a result it is necessary to use several sets of approximate factors to adequately cover the speed range and the types of vehicles.

This Appendix contains a summary of the best approximations currently known. In several high speed cases the conventional airplane factors of Refs. 44 are used. Most of the factors unique to helicopters were taken from Refs. 2 and 45.

The conditions below list the configurations and forward speeds which were used to check the validity of the approximate factors. In all cases

Vehicle	U_o (ft/sec)
H-19 single-rotor helicopter	1 50 70 115
HUP-1 tandem-rotor helicopter	1 40 80 120

the approximate factors were within 5 percent of the exact values. The general forms of the transfer functions are given in Table A-II. Tables A-II and A-V list the denominator approximate factors for the single rotor and tandem rotor helicopters. Tables A-III to A-IV list longitudinal and Tables A-VI to A-VII list the lateral approximate factors for the numerators of the single and tandem rotor helicopters. All the approximate factors

are for straight and level flight and for stability axis derivatives. If the conditions of validity are met, the approximate factors should generally be accurate within ±10 percent of the magnitude of the root, i.e., for a second-order pair the errors in the real and imaginary parts of the roots should be less than 10 percent of their frequency.

TABLE A-I

TRANSFER FUNCTION FORMS

GENERAL:

$$\frac{\lambda(s)}{\delta(s)} = \frac{N_\lambda(s)}{\Delta(s)}$$

LONGITUDINAL:

$$\Delta(s) = \left(s^2 + 2\zeta_p\omega_p s + \omega_p^2\right)\underbrace{\left(s^2 + 2\zeta_{sp}\omega_{sp}s + \omega_{sp}^2\right)}_{\text{or}}$$

$$\left(s + \frac{1}{T_{sp_1}}\right)\left(s + \frac{1}{T_{sp_2}}\right)$$

$$N_\theta(s) = \kappa_\theta\left(s + \frac{1}{T_{\theta_1}}\right)\left(s + \frac{1}{T_{\theta_2}}\right)$$

$$N_w(s) = \kappa_w\left(s + \frac{1}{T_{w_1}}\right)\underbrace{\left(s + \frac{1}{T_{w_2}}\right)\left(s + \frac{1}{T_{w_3}}\right)}_{\text{or}}$$

$$\left(s^2 + 2\zeta_w\omega_w s + \omega_w^2\right)$$

$$N_u(s) = \kappa_u\left(s + \frac{1}{T_{u_1}}\right)\underbrace{\left(s + \frac{1}{T_{u_2}}\right)\left(s + \frac{1}{T_{u_3}}\right)}_{\text{or}}$$

$$\left(s^2 + 2\zeta_u\omega_u s + \omega_u^2\right)$$

$$N_{\dot{h}}(s) = \kappa_h\left(s + \frac{1}{T_{h_1}}\right)\underbrace{\left(s + \frac{1}{T_{h_2}}\right)\left(s + \frac{1}{T_{h_3}}\right)}_{\text{or}}$$

$$\left(s^2 + 2\zeta_h\omega_h s + \omega_h^2\right)$$

LATERAL:

$$\Delta(s) = \left(s + \frac{1}{T_s}\right)\left(s + \frac{1}{T_R}\right)\left(s^2 + 2\zeta_d\omega_d s + \omega_d^2\right)$$

$$N_\varphi(s) = \kappa_\varphi\underbrace{\left(s^2 + 2\zeta_\varphi\omega_\varphi s + \omega_\varphi^2\right)}_{\text{or}}$$

$$\left(s + \frac{1}{T_{\varphi_1}}\right)\left(s + \frac{1}{T_{\varphi_2}}\right)$$

$$N_r(s) = \kappa_r\left(s + \frac{1}{T_r}\right)\left(s^2 + 2\zeta_r\omega_r s + \omega_r^2\right)$$

$$N_v(s) = \kappa_v\left(s + \frac{1}{T_{v_1}}\right)\underbrace{\left(s + \frac{1}{T_{v_2}}\right)\left(s + \frac{1}{T_{v_3}}\right)}_{\text{or}}$$

$$\left(s^2 + 2\zeta_v\omega_v s + \omega_v^2\right)$$

TABLE A-XI

HELICOPTER APPROXIMATE LONGITUDINAL DENOMINATOR TRANSFER FUNCTIONS

SINGLE ROTOR AND TANDEM ROTOR AT HOVER

FIRST COEFF.	APPROXIMATE FACTORS	CONDITIONS OF VALIDITY
1	$\frac{1}{T_{sp_1}} \doteq -Z_w$ $\frac{1}{T_{sp_2}} \doteq \frac{-3M_q}{4} + \sqrt{\frac{M_q^2}{16} - \frac{gM_u}{2M_q}}$ $2\zeta_p\omega_p \doteq -X_u - \frac{M_q}{4} - \sqrt{\frac{M_q^2}{16} - \frac{gM_u}{2M_q}}$ $\omega_p^2 \doteq \frac{gM_u}{\frac{-3M_q}{4} + \sqrt{\frac{M_q^2}{16} - \frac{gM_u}{2M_q}}}$	$\lvert M_q \rvert \gg \lvert X_u \rvert$ $\left\lvert \frac{gM_u}{M_q^3} \right\rvert < 1$ If $\lvert X_u \rvert \gg \lvert M_q \rvert$ and $\left\lvert \frac{gM_u}{X_u^3} \right\rvert < 1$ interchange X_u and M_q in equations

SINGLE ROTOR FOR $U_o > 50$ fps

FIRST COEFF.	APPROXIMATE FACTORS	CONDITIONS OF VALIDITY
1	$\frac{1}{T_{sp_1}}\frac{1}{T_{sp_2}}$ or $\omega_{sp}^2 \doteq Z_wM_q - U_oM_w$ $\frac{1}{T_{sp_1}} + \frac{1}{T_{sp_2}}$ or $2\zeta_{sp}\omega_{sp} \doteq -Z_w - M_q - U_oM_{\dot{w}}$ $\frac{1}{T_{p_1}}\frac{1}{T_{p_2}}$ or $\omega_p^2 \doteq \frac{g(M_wZ_u - M_uZ_w)}{Z_wM_q - U_oM_w}$ $\frac{1}{T_{p_1}} + \frac{1}{T_{p_2}}$ or $2\zeta_p\omega_p \doteq -X_u - \frac{M_u(X_wU_o - g)}{Z_wM_q - U_oM_w}$	ω_{sp}^2 or $\left\lvert \frac{1}{T_{sp_1}} \cdot \frac{1}{T_{sp_2}} \right\rvert > 4\omega_p^2$ or $4\left\lvert \frac{1}{T_{p_1}} \cdot \frac{1}{T_{p_2}} \right\rvert$ $\lvert X_u \rvert \ll \lvert M_q + Z_w + U_oM_{\dot{w}} \rvert$ $\lvert X_u(M_q + Z_w + U_oM_{\dot{w}}) \rvert \ll \lvert M_qZ_w - U_oM_w \rvert$ $\lvert Z_u(gM_{\dot{w}} + X_wM_q) \rvert \ll \lvert M_u(g - U_oX_w) - X_u(Z_wM_q - U_oM_w) \rvert$

TANDEM ROTOR FOR $U_o > 40$ fps

FIRST COEFF.	APPROXIMATE FACTORS	CONDITIONS OF VALIDITY
1	$\frac{1}{T_{sp_1}} \doteq -\left(\frac{M_q + Z_w + U_oM_{\dot{w}}}{2}\right) + \sqrt{\left(\frac{M_q + Z_w + U_oM_{\dot{w}}}{2}\right)^2 + U_oM_w - M_qZ_w}$ $\frac{-1}{T_{sp_2}} \doteq \omega_p \doteq -\left[gT_{sp_1}(M_wZ_u - M_uZ_w)\right]^{1/3}$ $2\zeta_p\omega_p \doteq -X_u - \frac{1}{T_{sp_2}}$	$\left\lvert \frac{1}{T_{sp_2}} \right\rvert > 3\left\lvert \frac{1}{T_{sp_1}} \right\rvert$ $\lvert gM_u \rvert \gg \lvert -X_u(M_qZ_w - U_oM_w) - M_uU_oX_w + M_qX_wZ_u + gX_uM_{\dot{w}} \rvert$ If M_u is too small to satisfy this condition, use AZ

TABLE A-III

SINGLE ROTOR HELICOPTER APPROXIMATE LONGITUDINAL NUMERATOR TRANSFER FUNCTIONS

QUAN.	FIRST COEFF.	APPROXIMATE FACTORS	VALIDITY CONDITIONS
LONGITUDINAL CYCLIC PITCH			
θ	M_{B_1}	$\frac{1}{T_{\theta_1}} \doteq -X_u + M_u \frac{X_{B_1}}{M_{B_1}}$ $\frac{1}{T_{\theta_2}} \doteq -Z_w$	$\left\lvert\frac{1}{T_{\theta_1}}\right\rvert \ll \left\lvert\frac{1}{T_{\theta_2}}\right\rvert$ $M_w Z_u \ll Z_w M_u$ $X_w Z_u \ll Z_w X_u$
w	Z_{B_1}	$\frac{1}{T_w} \doteq \frac{M_{B_1}}{Z_{B_1}} U_o$ $\omega_w^2 \doteq \frac{g}{U_o M_{B_1}} (Z_u M_{B_1} - M_u Z_{B_1})$ $2\zeta_w \omega_w \doteq \frac{1}{U_o M_{B_1}}\left[X_\delta (M_u U_o - Z_u M_q) + Z_{B_1} X_u M_q - M_{B_1} U_o X_u - \omega_w^2 Z_\delta\right]$	$U_o \neq 0$ (at $U_o = 0$, $Z_{B_1} = 0$) $\left\lvert\frac{1}{T_w}\right\rvert \gg \omega_w$
u	X_{B_1}	$\frac{1}{T_u} \doteq -Z_w$ $\omega_u^2 \doteq \frac{-g}{X_{B_1}}\left(M_{B_1} - Z_{B_1}\frac{M_w}{Z_w}\right) \doteq -M_{B_1} + Z_{B_1}\frac{M_w}{Z_w}$ $2\zeta_u \omega_u \doteq -M_q$	$\lvert M_q + Z_w\rvert \gg \left\lvert U_o M_{\dot{w}} - \frac{Z_\delta X_w}{X_{B_1}}\right\rvert$ $\lvert X_{B_1} Z_w M_q - g M_\delta\rvert \gg \lvert U_o (X_w M_\delta - X_\delta M_w) - Z_\delta (g M_{\dot{w}} + X_w M_q)\rvert$
h	$-Z_{B_1}$	$\frac{1}{T_h} \doteq -X_u + Z_u \frac{X_{B_1}}{Z_{B_1}}$ $\omega_h^2 \doteq -M_{B_1} - U_o M_w$ $2\zeta_h \omega_h \doteq -M_q$	$g - U_o X_w \doteq X_{B_1}$ $-U_o Z_w \doteq Z_{B_1}$ $U_o \neq 0$ (at $U_o = 0$, $Z_{B_1} = 0$)
COLLECTIVE PITCH			
θ	M_c	$\frac{1}{T_{\theta_1}} \doteq -X_u - \frac{M_u (Z_w X_c - X_w Z_c)}{M_w Z_c - Z_w M_c}$ $\frac{1}{T_{\theta_2}} \doteq -Z_w + \frac{M_w Z_c + M_u X_c}{M_c} \doteq -Z_w + \frac{M_w}{M_c} Z_c$	$\left\lvert\frac{1}{T_{\theta_1}}\right\rvert \ll \lvert -Z_w\rvert$ $\left\lvert\frac{1}{T_{\theta_1}}\right\rvert \ll \left\lvert\frac{1}{T_{\theta_2}}\right\rvert$ $M_c \neq 0$ (Note: $M_c = 0$ at $U_o = 0$)
w	Z_c	$\frac{1}{T_{w_1}} \doteq -M_q$ $\frac{1}{T_{w_2}} \frac{1}{T_{w_3}}$ or $\omega_w^2 \doteq \frac{g M_u}{-M_q}$ $-0.5 < \zeta < 0$	$U_o \neq 0$ [for $U_o = 0$, $w(s)/c(s) = Z_c/s - Z_w$] $\lvert X_u\rvert \ll \lvert 1/T_w\rvert$ $\lvert X_u\rvert \ll \lvert\omega_w\rvert$ $\lvert X_c\rvert \ll Z_c$ $\lvert M_q\rvert \gg \lvert U_o M_c / Z_c\rvert$ No simple approximate formula for ζ has been found
u	X_c	$\frac{1}{T_u} \doteq -Z_w + X_w \frac{Z_c}{X_c}$ $\omega_u^2 \doteq \frac{-g M_c}{X_c}$ $2\zeta_u \omega_u \doteq -M_q$	$X_c \neq 0$ (Note: $X_c = 0$ at $U_o = 0$) $\lvert X_w Z_c\rvert \ll \lvert Z_w X_c\rvert$
h	$-Z_c$	For $U_o \neq 0$, $1/T_h$, ζ_h, and ω_h are given by factors of $s^3 - M_q s^2 - \left(U_o M_w + \frac{M_c Z_w U_o}{Z_c}\right)s - M_u (g - U_o X_w) = 0$ No simple literal factors have been found for this cubic	For $U_o = 0$, $\frac{h}{c}(s) = \frac{-Z_c}{s(s - Z_w)}$

TABLE A-IV

TANDEM ROTOR HELICOPTER APPROXIMATE LONGITUDINAL NUMERATOR TRANSFER FUNCTIONS

QUAN.	FIRST COEFF.	APPROXIMATE FACTORS	VALIDITY CONDITIONS
DIFFERENTIAL COLLECTIVE PITCH (PLUS SWASHPLATE TILT)			
θ	$M_B + Z_B M_{\dot{w}}$	$\frac{1}{T_{\theta_1}} \doteq \frac{X_B(Z_u M_w - M_u Z_w) - Z_B(X_u M_w - M_u X_w) + M_B(X_u Z_w - Z_u X_w)}{-Z_w(M_B + Z_B M_{\dot{w}})}$ $\frac{1}{T_{\theta_2}} \doteq -Z_w$	$\left\lvert\frac{1}{T_{\theta_1}}\right\rvert \ll \left\lvert\frac{1}{T_{\theta_2}}\right\rvert$
w	Z_B	$\frac{1}{T_{w_1}} \doteq U_0 \frac{M_B}{Z_B}$ $2\zeta_w\omega_w$ or $\frac{1}{T_{w_2}} + \frac{1}{T_{w_3}} \doteq \frac{Z_B}{U_0 M_B}\left[X_B(U_0 M_u - Z_u M_q) + Z_B X_u M_q - M_B U_0 X_u - \omega_w^2\right]$ ω_w^2 or $\frac{1}{T_{w_2}}\frac{1}{T_{w_3}} \doteq \frac{g}{U_0}\left(-Z_u + \frac{M_u}{M_B} Z_B\right)$	$\left\lvert\frac{1}{T_{w_1}}\right\rvert \gg \omega_w^2$, $\lvert U_0 M_B\rvert \gg X_u Z_B$ $\left\lvert\frac{1}{T_{w_1}}\right\rvert \gg \lvert M_q\rvert$, $U_0 \neq 0$
u	X_B	$\frac{1}{T_u} \doteq -Z_w$ $\omega_u^2 \doteq \frac{-g}{X_B}\left(M_B - Z_B \frac{M_w}{Z_w}\right)$ $2\zeta_u\omega_u \doteq -M_q$	$\lvert Z_w\rvert \gg \lvert M_w\rvert$ $\lvert Z_w\rvert \gg \lvert X_w\rvert$
h	$-Z_B$	$\frac{1}{T_h} \doteq \frac{Z_B}{M_B Z_w U_0}\left\{U_0 X_u(Z_w M_B - M_w Z_B) - (g - X_q)(Z_u M_B - M_u Z_B) + U_0 X_B(Z_u M_w - M_u Z_w)\right\}$ $2\zeta_h\omega_h \doteq -M_q - U_0 M_{\dot{w}}$ $\omega_h^2 \doteq \frac{M_B U_0 Z_w}{Z_B}$	$U_0 \neq 0$ $\lvert U_0 X_B(M_w Z_u - M_u Z_w) + g(Z_u M_B - M_u Z_B)\rvert$ $\gg \lvert U_0 Z_B(M_u X_w - M_w X_u) + U_0 M_B(Z_w X_u - Z_u X_w)\rvert$ $\frac{M_w Z_B}{Z_w} \ll M_B$
COLLECTIVE PITCH			
θ	$Z_c M_{\dot{w}}$	$\frac{1}{T_{\theta_1}} \doteq \frac{-M_w}{M_{\dot{w}}}$ $\frac{1}{T_{\theta_2}} \doteq -X_u + M_u \frac{X_w}{M_w}$	$M_{\dot{w}} \neq 0$ $\lvert M_c\rvert \ll \lvert Z_c\rvert$ $\lvert X_c\rvert \ll \lvert Z_c\rvert$ $U_0 \neq 0$
w	Z_c	$\frac{1}{T_{w_1}} \doteq -M_q$ ω_w^2 or $\frac{1}{T_{w_2}}\frac{1}{T_{w_3}} \doteq \frac{gM_u}{-M_q}$ $2\zeta_w\omega_w$ or $\frac{1}{T_{w_2}} + \frac{1}{T_{w_3}} \doteq -X_u - \frac{gM_u}{(M_q)^2}$	$M_q \gg X_u$ $X_u \ll 1$
u	$Z_c X_w$	$\frac{1}{T_{u_1}}\frac{1}{T_{u_2}} \doteq \frac{-gM_w}{X_w}$ $\frac{1}{T_{u_1}} + \frac{1}{T_{u_2}} \doteq -M_q - \frac{gM_{\dot{w}}}{X_w}$	$X_c = M_c = 0$ $U_0 \neq 0$
h	$-Z_c$	$\frac{1}{T_{h_1}} \doteq X_u + (g - U_0 X_w)\frac{M_u}{U_0 M_w}$ $\frac{1}{T_{h_2}} + \frac{1}{T_{h_3}}$ or $2\zeta_h\omega_h \doteq -M_q$ $\frac{1}{T_{h_2}}\frac{1}{T_{h_3}}$ or $\omega_h^2 \doteq -U_0 M_w$	$U_0 \neq 0$ $\lvert Z_u\rvert \ll \lvert U_0 Z_w\rvert$ $\lvert M_u\rvert \ll \lvert U_0 M_w\rvert$ $X_c = M_c = 0$

TABLE A-V

HELICOPTER APPROXIMATE LATERAL DENOMINATOR TRANSFER FUNCTIONS

SINGLE ROTOR FOR U > 0

FIRST COEFF.	APPROXIMATE FACTORS	CONDITIONS OF VALIDITY
1	$\omega_d^2 \doteq U_0 N'_v$ $2\zeta_d\omega_d \doteq -(Y_v + N'_r) - \frac{L'_v}{N'_v}\left(N'_p - \frac{g}{U_0}\right)$ $\frac{1}{T_R} \doteq -L'_p$ $\frac{1}{T_s} \doteq -\frac{g}{U_0 L'_p}\left(\frac{L'_v}{N'_v} N'_r - L'_r\right)$	$U_0 \neq 0$ $N_v \neq 0$ $\left\lvert\frac{1}{T_s}\right\rvert \ll \left\lvert\frac{1}{T_R}\right\rvert$ $Y_r \doteq Y_p \doteq 0$ $\left\lvert\frac{L'_v g}{U_0}\right\rvert \ll L'^2_p$ $\lvert g L'_v\rvert \ll \lvert L'^3_p\rvert$

TANDEM ROTOR

1	$\frac{1}{T_s} \doteq -N'_r\left[1 + \frac{U_0}{g}\left(N'_p - \frac{N'_v L'_p}{L'_v}\right) - \frac{L'_r N'_v}{N'_r L'_v}\right]$ $\frac{1}{T_R} \doteq -L'_p - \frac{gL'_v}{L'^2_p}$ $2\zeta_d\omega_d \doteq -Y_v + \frac{gL'_v}{L'^2_p}$ In hover, a better approximation is: $2\zeta_d\omega_d \doteq -Y_v - \frac{L'_p}{4} - \sqrt{\frac{L'^2_p}{16} + \frac{gL'_v}{2L'_p}}$ $\omega_d^2 \doteq \frac{-gL'_v + U_0(L'_v N'_p - N'_v L'_p)}{-L'_p - \frac{gL'_v}{L'^2_p}}$	

TABLE A-VI

SINGLE ROTOR HELICOPTER APPROXIMATE LATERAL NUMERATOR TRANSFER FUNCTIONS

QUANTITY	FIRST COEFF.	APPROXIMATE FACTORS	VALIDITY CONDITIONS
LATERAL CYCLIC PITCH (AILERON)			
φ	L_δ'	$2\zeta_\varphi\omega_\varphi \doteq -Y_v - N_r'$ $\omega_\varphi^2 \doteq U_0 N_v'$	$\lvert Y_\delta L_r'\rvert \ll \lvert U_0 L_\delta'\rvert$ $(-Y_v - N_r')^2 < 4U_0N_v'$ For $U_0 \doteq 0$, use exact expression
r	$Y_\delta N_v' + L_\delta' N_p'$	$2\zeta_r\omega_r \doteq \dfrac{Y_\delta(L_v'N_r' - N_v'L_p') - L_\delta' Y_v N_p'}{Y_\delta N_v' + L_\delta' N_p'}$ $\omega_r^2 \doteq g\dfrac{L_\delta' N_v'}{L_\delta' N_p' + Y_\delta N_v'}$	$N_\delta = 0$ Expressions are exact for $N_\delta = 0$ $N_\delta \doteq 0$ introduces third root and changes first coefficient to N_δ
v	Y_δ	Simplest generally valid forms are $2\zeta_v\omega_v + \frac{1}{T_v} \doteq -L_p' - N_r'$ $\omega_v^2 + 2\zeta_v\omega_v\frac{1}{T_v} \doteq N_r'L_p' - \frac{L_\delta'}{Y_\delta}(g - U_0N_p')$ $\omega_v^2\frac{1}{T_v} \doteq g\frac{L_\delta'}{Y_\delta}N_r'$	$N_\delta \doteq 0$ $L_r'N_p' \ll L_p'N_r'$
TAIL ROTOR COLLECTIVE PITCH (RUDDER)			
φ	L_δ'	$\frac{1}{T_{\varphi_1}} + \frac{1}{T_{\varphi_2}} \doteq -Y_v - N_r' + \frac{N_\delta'}{L_\delta'}L_r'$ $\frac{1}{T_{\varphi_1}}\frac{1}{T_{\varphi_2}} \doteq U_0N_v' + \frac{Y_\delta}{L_\delta'}L_v'N_r' - \frac{N_\delta'}{L_\delta'}U_0L_v'$	$\lvert Y_\delta L_v'\rvert \ll \lvert N_\delta' L_r' + L_\delta'(Y_v + N_r')\rvert$ $1/T_{\varphi_1} + 1/T_{\varphi_2}$ is given by small differences between $-Y_v - N_r'$ and $(N_\delta/L_\delta)L_r'$; hence, accuracy is low
r	N_δ'	$\frac{1}{T_r} \doteq -L_p'$ $\omega_r^2 \doteq -\frac{g}{L_p'}\left(-L_v' + \frac{L_\delta'}{N_\delta'}N_v'\right)$ $2\zeta_r\omega_r \doteq 0$	$\lvert L_p'\rvert \gg \frac{L_\delta'}{N_\delta'}N_p' + \frac{Y_\delta}{N_\delta'}N_v'$ ζ is typically of the order of 0.02 to 0.05, but no simple approximation for ζ has been found
v	Y_δ	$\frac{1}{T_{v_1}} \doteq 0$ $\frac{1}{T_{v_2}} \doteq -\left(L_p' + U_0\frac{N_\delta'}{Y_\delta}\right)$ $\frac{1}{T_{v_3}} \doteq T_{v_2}\left[L_p'N_r' - L_r'N_p' + \frac{L_\delta'}{Y_\delta}(U_0N_p' - g) + \frac{N_\delta'}{Y_\delta}U_0L_p'\right]$	$U_0 \neq 0$, $U_0 \doteq 0$ $\frac{1}{T_{v_1}} \ll \frac{1}{T_{v_2}}$ $\frac{1}{T_{v_1}}$ is of the order of 0.002 typically $\frac{1}{T_{v_2}}$ is of the order of 50 typically

TABLE A-VII

TANDEM ROTOR HELICOPTER APPROXIMATE LATERAL NUMERATOR TRANSFER FUNCTIONS

QUANTITY	FIRST COEFF.	APPROXIMATE FACTORS	VALIDITY CONDITIONS
LATERAL CYCLIC PITCH			
φ	L_δ	$\left.\begin{array}{l}\frac{1}{T_{\varphi_1}T_{\varphi_2}} \\ \text{or } \omega_\varphi^2\end{array}\right\} = U_0N_v + Y_vN_r$ $\left.\begin{array}{l}\text{or } 2\zeta_\varphi\omega_\varphi \\ \frac{1}{T_{\varphi_1}} + \frac{1}{T_{\varphi_2}}\end{array}\right\} = -Y_v - N_r$	$N_\delta I_{xz} \ll L_\delta I_x$, $U_0 \neq 0$ $\left\|\frac{Y_\delta}{L_\delta}\left(L_v + \frac{I_{xz}}{I_x}N_v\right) + \frac{N_\delta}{L_\delta}\left(L_r - \frac{I_{xz}}{I_x}Y_v\right)\right\| \ll \|N_r + Y_v\|$ $\|Y_\delta(L_rN_v + L_vN_r) - N_\delta(U_0L_v + Y_vL_r)\|$ $\ll L_\delta(U_0N_v + Y_vN_r)$ For $U_0 \doteq 0$, use $1/T_{\varphi_1} \doteq -N_r$ $1/T_{\varphi_2} \doteq -Y_v + (Y_\delta/L_\delta)L_v$
r	N_δ'	$\frac{1}{T_r} \doteq \left[g\left(-L_v' + N_v'\frac{L_\delta'}{N_\delta'}\right)\right]^{1/3}$, $\omega_r = \left\|\frac{1}{T_r}\right\|$ $\zeta_r \doteq 0.5$ for $\frac{1}{T_r} < 0$ $\zeta_r \doteq -0.5$ for $\frac{1}{T_r} > 0$	$\frac{L_p}{N_p} \doteq \frac{L_{\delta_a}}{N_{\delta_a}}$
v	Y_δ	$\frac{1}{T_{v_1}} \doteq -N_r$ $2\zeta_v\omega_v$ or $\frac{1}{T_{v_2}} + \frac{1}{T_{v_3}} \doteq -L_p' + \frac{U_0N_\delta'}{Y_\delta}$ ω_v^2 or $\frac{1}{T_{v_2}}\frac{1}{T_{v_3}} \doteq -\frac{L_\delta'N_r'}{1/T_{v_1}}$	$g \doteq Y_{\delta_a}$ $\|L_\delta'N_r'\| \gg \|N_\delta'L_p'\|$
DIFFERENTIAL LATERAL CYCLIC PITCH (RUDDER)			
φ	L_δ'	$\omega_\varphi^2 \doteq U_0\left(N_v' - \frac{N_\delta'}{L_\delta'}L_v'\right)$ $2\zeta_\varphi\omega_\varphi \doteq \frac{N_\delta'}{L_\delta'}L_r' - Y_v - N_r'$	$U_0 \neq 0$ (Note rapid change in nature of factors with U_0, for U_0 small) $Y_{\delta_r} \doteq 0$ For $U_0 = 0$, see text
r	N_δ'	$\frac{1}{T_r} \doteq -L_p'$ $\omega_r^2 \doteq \frac{-g}{L_p'}\left(-L_v' + N_v'\frac{L_\delta'}{N_\delta'}\right)$ $2\zeta_r\omega_r \doteq -Y_v + \frac{\omega_r^2}{L_p'}$	$Y_{\delta_r} \doteq 0$ $\left\|\frac{1}{T_r}\right\| \gg \omega_r$ $\left\|\frac{1}{T_r}\right\| \gg \|2\zeta_r\omega_r\|$; $-L_p' \gg \left\|N_p'\frac{N_\delta'}{N_\delta'}\right\|$
v	$-U_0N_\delta'$	$\frac{1}{T_1} + \frac{1}{T_2} \doteq -\left[L_p' + \frac{L_\delta'}{N_\delta'}\left(\frac{g}{U_0} - N_p'\right)\right]$ $\frac{1}{T_1}\frac{1}{T_2} \doteq -\left[\frac{g}{U_0}\left(L_r' - \frac{L_\delta'}{N_\delta'}N_r'\right)\right]$	$Y_\delta = 0$, $U_0 \neq 0$ (but valid for $U_0 \doteq 0$) At $U_0 = 0$, numerator becomes first-order $g\left[L_\delta'(s - N_r') + L_r'N_\delta'\right]$

APPENDIX B

METHOD OF APPROACH FOR CLOSED-LOOP ANALYSES

I. HOVER OVER A SPOT WITH GUST DISTURBANCES

Hovering in gusty air, the pilot is concerned with the variations in position, attitude, and control deflection as influenced by the gust disturbances. It is the purpose of this Appendix to present the approach used to examine the effects of random u-gust inputs on pilot closures and to calculate the effects of changes in the stability derivatives on the rms (root mean square) x, θ, and $M_{\delta_B}\delta_B$ responses with both the θ- and x-loops closed. In particular, variations in the pilot gains, leads, and time delay from the nominal values of Section II will be shown. The nominal gains will be shown to be reasonable in terms of minimizing the composite rms responses in θ, x, and $M_{\delta_B}\delta_B$. Likewise the various stability derivative effects will be analyzed to determine those parameters which have the most influence on the attitude and position responses to u-gust and on the control power requirements for hovering in gusty air.

The five stability derivatives, X_u, X_{δ_B}, M_u, M_q, M_{δ_B}, completely specify the longitudinal dynamics (excluding the plunging mode) of a hovering vehicle. In the handling qualities analyses which follow, the effects of control sensitivity, M_{δ_B}, will not be considered. The analysis method considers only the pilot-vehicle transfer function pole-zero locations and total loop gain (product of pilot gain and control sensitivity). The resulting conclusions are therefore only valid for situations in which M_{δ_B} is adjusted to its optimum value for the selected values of the other terms. In other words, degradations in pilot rating due to too high or too low a control sensitivity are not considered here.

Eliminating M_{δ_B}, we will consider the four quantities:

$$X_u \quad , \quad X_{\delta_B}/M_{\delta_B} \quad , \quad M_u \quad , \quad M_q$$

Of these, M_u and M_q are generally recognized as the most important. Consequently, the major emphasis here will be on the effects of these two derivatives; that X_u and $X_{\delta_B}/M_{\delta_B}$ are of secondary importance will be demonstrated. The main point of the analysis is an examination of four combinations of derivatives; two values of both M_u and M_q for set values of X_u and $X_{\delta_B}/M_{\delta_B}$. These values of M_u and M_q were selected to bracket the most critical region ever to be expected with the helicopter.

The stability derivatives for our representative sampling of helicopter vehicles are given in Tables B-I and B-II. From this survey the following values were selected:

$$X_u(Y_v) = -0.13 \text{ sec}^{-1}$$

$$X_{\delta_B}/M_{\delta_B}(Y_{\delta_A}/L_{\delta_A}) = 0$$

$$M_u(-L_v) = \begin{cases} 0.0088 \\ 0.088 \end{cases} (\text{ft-sec})^{-1}$$

$$M_q(L_p) = \begin{cases} -0.15 \\ -1.5 \end{cases} \text{sec}^{-1}$$

The root positions for the hover cubic are shown in Fig. B-1 which also includes the root locations for the surveyed vehicles. It can be seen that the four values selected adequately cover the range of critical root positions. For our present sampling none of the vehicle root positions lie to the right or above the most critical boundaries.

Figure B-1 also illustrates the effects of M_u and M_q on the characteristic roots. Increasing M_u increases the phugoid frequency at nearly constant damping ratio and increases $1/T_{sp}$. Increasing the pitch damping (M_q more negative) increases the phugoid damping at roughly constant frequency and also increases $1/T_{sp}$.

Making X_u more negative (not shown in Fig. B-1) also increases phugoid damping at roughly constant damped frequency $\omega_p\sqrt{1-\zeta_p^2}$ and increases $1/T_{sp}$. The increases in $\zeta_p\omega_p$ and $1/T_{sp}$ are approximately equal to one-third the change in X_u (see Appendix A).

TABLE B-I

SURVEY OF LONGITUDINAL HOVER DERIVATIVES

VEHICLE WT AND I_y LB SLUG-FT^2	M_u (FT-SEC)$^{-1}$	$-M_q$ SEC^{-1}	$-X_u$ SEC^{-1}	$-M_{\delta_B}$ (SEC^2-IN)$^{-1}$	$-\frac{X_{\delta_B}}{M_{\delta_B}}$ FT
H-19 SINGLE-ROTOR 6,400 9,640	.0061	.61	.03	.20	4.8
S-58 SINGLE-ROTOR 11,600 27,500	.005	.60	.03*	.18	5.3
HUP-1 TANDEM ROTOR 5,430 10,080	.035	2.0	.019	.41	1.9
VERTOL 107 TANDEM ROTOR 13,000 75,000	.00082	1.34	.018	.312	-.32

TABLE B-II

SURVEY OF LATERAL HOVER DERIVATIVES

VEHICLE WT AND I_x LB SLUG-FT^2	$-L_v$ (FT-SEC)$^{-1}$	$-L_p$ SEC^{-1}	$-Y_v$ SEC^{-1}	L_{δ_A} (SEC^2-IN)$^{-1}$	$\frac{Y_{\delta_A}}{L_{\delta_A}}$ FT
H-19 W 6,400 I_x 2,188	.052	3.18	.073	.87	1.1
HUP-1 W = 5,550 I_x = 971	.032	1.49	.028	.49	.82
S-58 W = 11,600 I_x = 5,895	.025	2.50	.03	.55	1.0*
VERTOL 107 W = 13,000 I_x = 9,200	.011	1.00	.023	.66	3.25

*Estimates - Not Given with Data Source

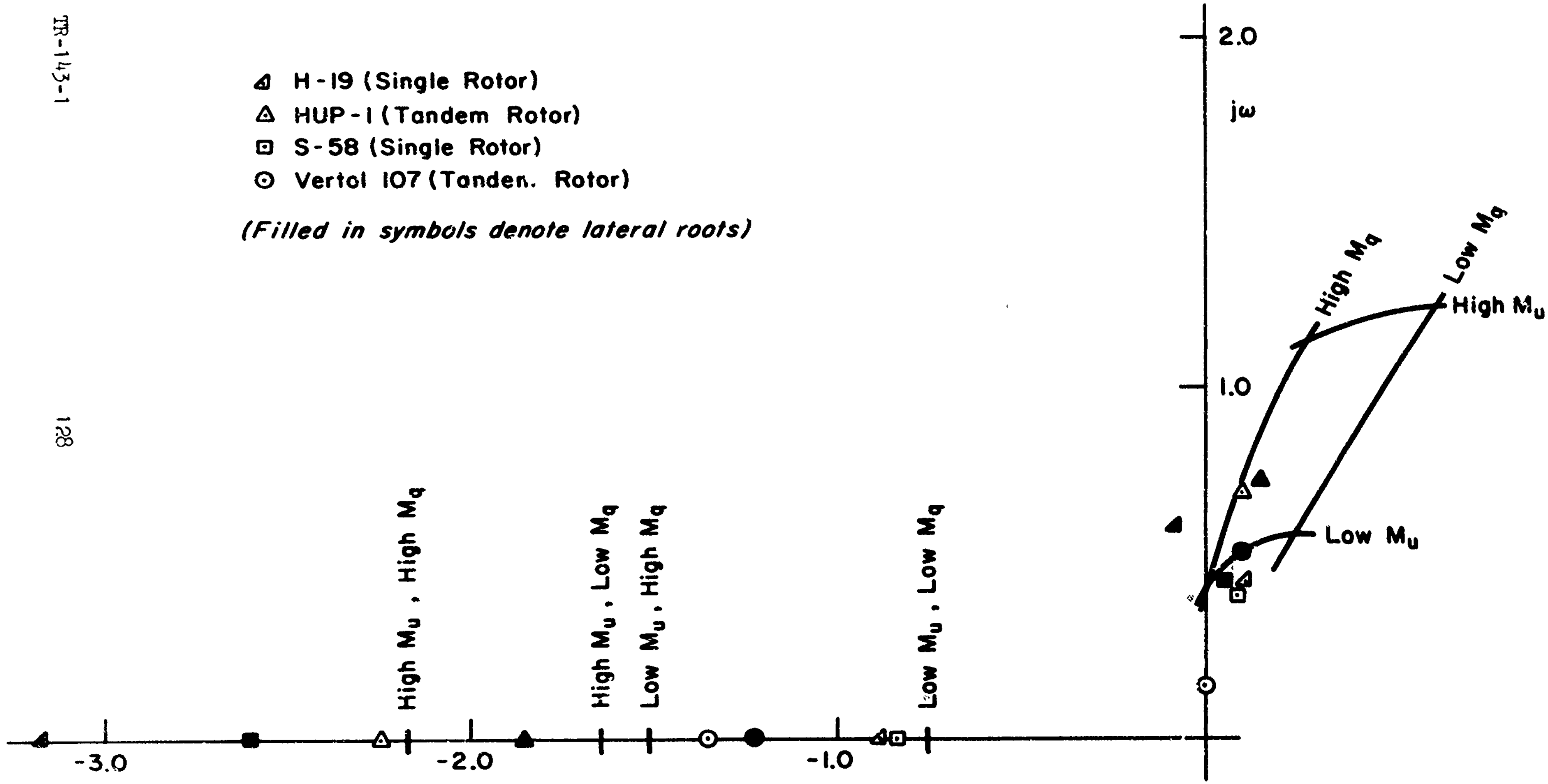


Figure B-1. Hover Root Position for Several Helicopters

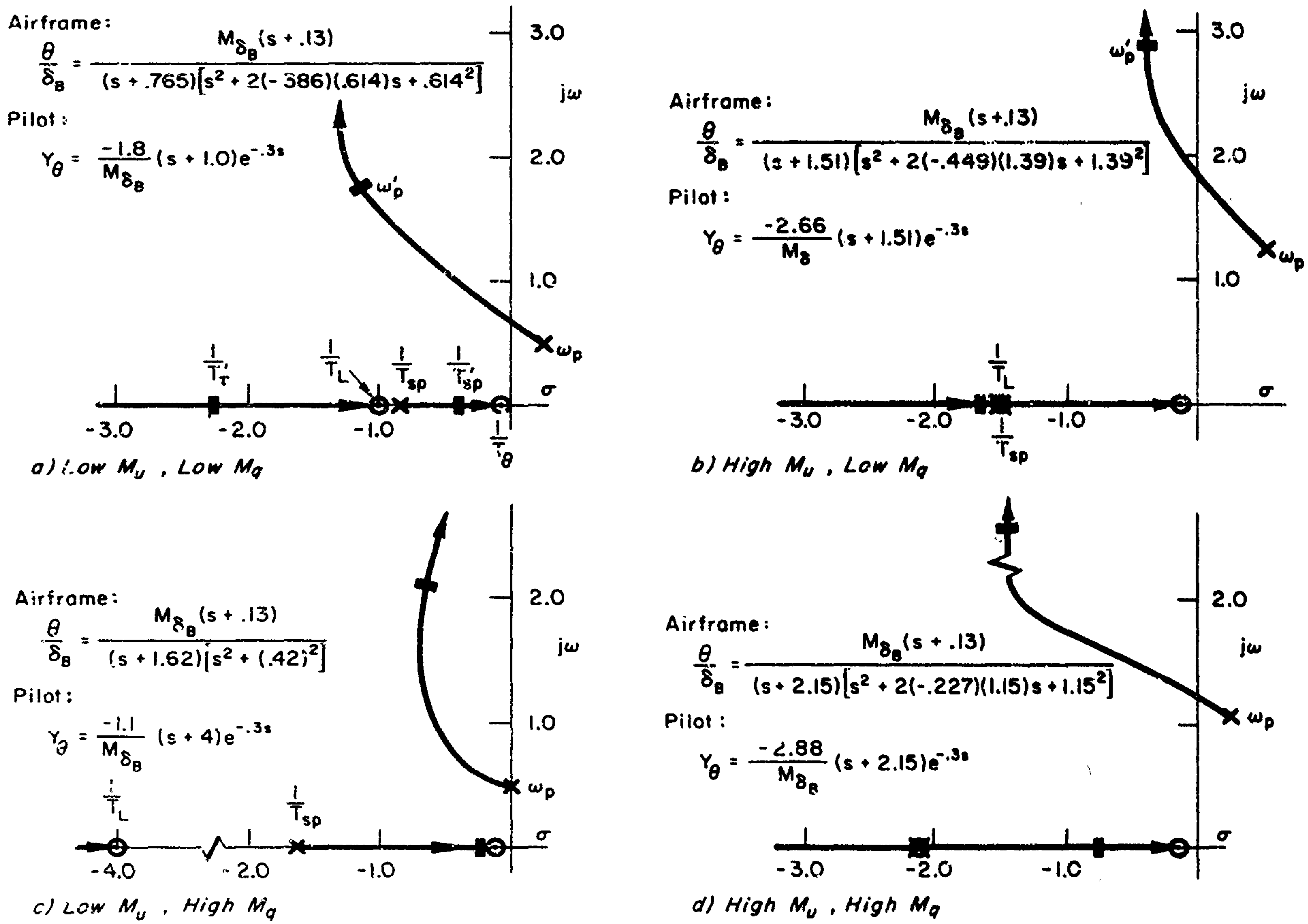


Figure B-2. Kover Pitch Loop Closures

Of course $X_{\delta_B}/M_{\delta_B}$ has no effect on the characteristic roots, but does influence the θ/δ_B and x/δ_B numerators.

Additional information on the effects of the derivatives on the open-loop dynamics is readily obtained via the approximate factors of Appendix A.

Pitch Closure

The pilot model selected for this loop was a properly placed lead and an effective transport lag of 0.3 sec. The pilot lead and gain are generally adjusted for a crossover frequency of 2-3 rad/sec with roughly 30 deg phase margin. The $\theta \rightarrow \delta_B$ closures for the four combinations of M_u and M_q are illustrated in Fig. B-2; the key parameters are listed in Table B-III.

TABLE B-III

ATTITUDE LOOP CLOSURE PARAMETERS

CASE		PHASE MARGIN	GAIN MARGIN	CROSS-OVER FREQ.	PILOT LEAD, T_L	CLOSED-LOOP ROOTS				HIGH FREQ. LOOP GAIN, κ_θ	D.C. LOOP GAIN, K_θ
						ζ_p'	ω_p'	$\left(\frac{1}{T_{sp}}\right)'$	$\frac{1}{\tau'}$		
M_u	M_q	deg	db	rad/sec	sec		rad/sec	sec^{-1}	sec^{-1}	rad/sec	
Low	Low	33	9	2.0	1.0	0.59	2.1	0.33	2.3	−1.80	0.825
High	Low	12	5	3.0	0.66	0.19	3.0	1.5	1.6	−2.66	0.184
Low	High	30	10	2.0	0.25	0.29	2.2	0.20	5.7	−1.10	2.02
High	High	26	6	3.2	0.46	0.33	3.8	0.77	2.2	−2.88	0.284

Note that for the high M_u, low M_q case the loop was closed at a lower phase margin than for the other cases. This was done after it was discovered that for this case a 30-deg phase margin closure resulted in gust

responses which were very sensitive to the θ-loop gain. Any condition which places tight restrictions on pilot gain is bad because the pilot cannot maintain his gain within narrow limits for extended periods. The pilot will want to operate in a broad optimum region where his errors are more or less minimum and his performance is not overly sensitive to his gain.

It can be seen from Table B-III that the most significant effect of increasing the pitch damping is to reduce the required pilot lead. This is as expected since the lead produces a pitching moment proportional to pitch rate (neglecting the lag introduced by the pilot's time delay). No serious effects of increasing M_u are noted for high M_q, although there is a substantial increase in $1/T'_{sp}$ (open-loop value was also increased).

Changing X_u shifts the open-loop parameters, $\zeta_p\omega_p$ and $1/T_{sp}$, by approximately one-third the change in X_u. Thus, realistic variations in X_u have little effect on the open-loop poles, but more strongly influence the θ-numerator zero, $1/T_{\theta_1}$, at $-X_u + M_uX_{\delta_B}/M_{\delta_B}$. As this zero is at a relatively low frequency, it has little effect on the closed-loop phugoid roots (see Fig. B-2). The effect on $1/T'_{sp}$ depends on the d.c. gain of the θ-loop. For high d.c. gain $1/T'_{sp}$ approaches the zero, so its change is nearly equal to the change in $-X_u$. For low d.c. gain $1/T'_{sp}$ is only slightly affected.

The θ-loop is affected by changes in $X_{\delta_B}/M_{\delta_B}$ only through the shift in the numerator zero at $-X_u + M_uX_{\delta_B}/M_{\delta_B}$. As the shift is proportional to M_u, the effect will be most important for large M_u cases. The high M_u, low M_q case was recalculated with a relatively large $X_{\delta_B}/M_{\delta_B}$ of -5 ft. The phase margin was increased from 12 to 20 deg and $1/T'_{sp}$ was reduced from 1.5 to 0.65 sec^{-1}. Neither of these changes is very important and all others are negligible.

Outer-Loop Position Closure

The analyses of the position control, $x \rightarrow \delta_B$, loops use the $\theta \rightarrow \delta_B$ closures of Table B-III as inner loops. As the x-loop can only be closed at relatively low frequencies, a pure gain pilot model is used. At

these low frequencies, pilot lead and transport lag tend to cancel and neither is significant in the crossover region of the $x \rightarrow \delta_B$ loop.

The key parameters for the $x \rightarrow \delta_B$ closures for the four M_u, M_q combinations are summarized in Table B-IV and the closures are illustrated in Fig. B-3. For ease of comparison a crossover frequency of roughly 0.3 rad/sec was used for all cases; this gives a minimum phase margin of about 30 deg.

TABLE B-IV

POSITION LOOP CLOSURE PARAMETERS

CASE		PHASE MARGIN	GAIN MARGIN	CROSS-OVER FREQ.	CLOSED-LOOP ROOTS					HIGH FREQ. LOOP GAIN, κ_x	LOW FREQ. LOOP GAIN, K_x
					ζ_x''	ω_x''	ζ_p''	ω_p''	$\left(\frac{1}{\tau}\right)''$		
M_u	M_q	deg	db	$\frac{\text{rad}}{\text{sec}}$		$\frac{\text{rad}}{\text{sec}}$		$\frac{\text{rad}}{\text{sec}}$	sec^{-1}	$\left(\frac{\text{rad}}{\text{sec}}\right)^4$	$\frac{\text{rad}}{\text{sec}}$
Low	Low	34	8	0.30	0.29	0.36	0.60	2.2	2.4	0.210	0.406
High	Low	70	17	0.30	0.85	0.61	0.21	3.0	2.0	1.04	0.311
Low	High	28	14	0.30	0.26	0.33	0.30	2.2	5.7	0.458	0.536
High	High	66	20	0.25	0.73	0.44	0.33	3.8	2.2	0.960	0.263

It can be seen from Table B-IV and Fig. B-3 that changing M_q has unimportant effects on the position loop. The dominant x-mode (ω_x'') is only slightly influenced by M_q. As before, increasing M_u has generally beneficial effects on the position loop. For the constant crossover frequency used, the phase and gain margins (also ζ_x'' and ω_x'') are increased; for given margins, the crossover frequency could have been increased. The beneficial effects are largely due to the increase in $1/T'_{sp}$ which accompanies an increase in M_u.

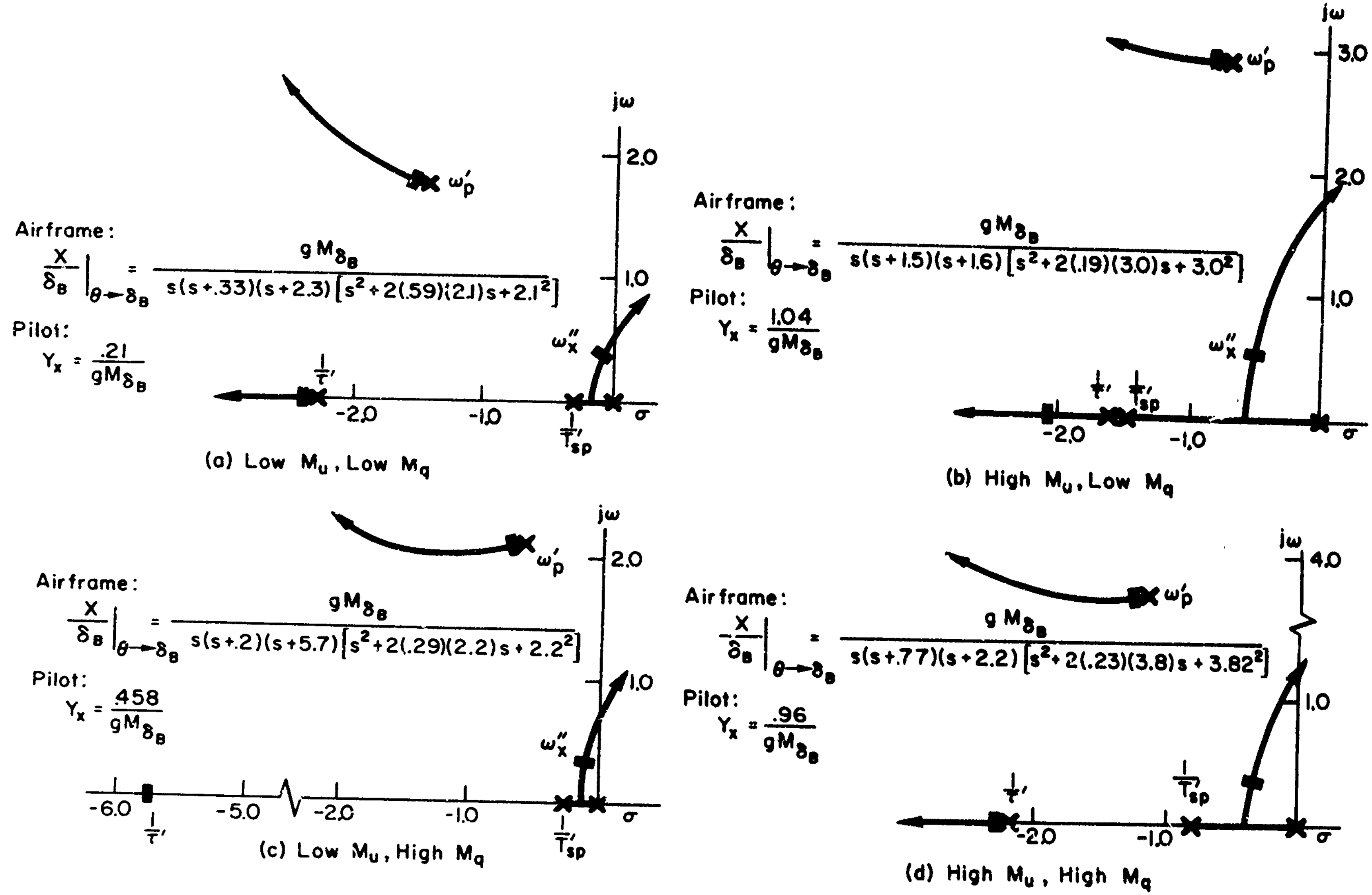


Figure B-3. Hover Position Loop Closures

Making X_u more negative has an advantageous effect on the position loop because of the increase in $1/T'_{sp}$. This allows a higher gain, higher bandwidth closure of the x-loop. However, for realistic variations in X_u the effect is rather unimportant.

A nonzero X_{δ_B} adds a second-order zero $\left[s^2 - M_q s - (gM_{\delta_B}/X_{\delta_B})\right]$ to the x/δ_B transfer function. For realistic values of $X_{\delta_B}/M_{\delta_B}$ this zero is at too high a frequency to be of significant benefit, e.g., for $X_{\delta_B}/M_{\delta_B} = -5$ ft the zero is at 2.54 rad/sec. For the high M_u, low M_q case this is more than offset by the lowered $1/T'_{sp}$ mentioned earlier. For a crossover frequency of 0.3 rad/sec this reduces the phase margin from 70 to 53 deg, reduces ζ''_x from 0.85 to 0.51, and ω''_x from 0.61 to 0.44 rad/sec. Thus, it appears that, for realistic values, X_{δ_B} has a negligible effect on position control if M_u is small and a detrimental effect if M_u is large. This fact is not only borne out by the above dynamic effect, but is discussed further in regard to control effects under gust responses on p. 142.

Gust Input Model

For analytical purposes it is desirable to have an input spectrum that is simple in form but which adequately represents the gust phenomena. Such a form, suggested and used in Ref. 45 and shown there to be adequate for the calculations of interest here, is the simple gust spectrum

$$\Phi_{\lambda g} = \frac{2\omega_g \sigma_{\lambda g}^2}{\omega^2 + \omega_g^2} \qquad \lambda = u, v, u \tag{B-1}$$

where $\sigma_{\lambda g}$ = rms gust velocity

ω_g = $(3/2)(V_{as}/L)$

V_{as} = steady-state airspeed (for hover, V_{as} = average wind speed)

L = integral scale of turbulence

The input spectra of Eq. B-1 are represented by the output of a linear filter whose input is white noise. The simple first-order filter model has the transfer function

$$Y_f = \frac{\sqrt{2\omega_g}\,\sigma_{\lambda g}}{s + \omega_g} \tag{B-2}$$

For the hover calculations the gust break frequency, $\omega_g = (3/2)(V_{as}/L)$, was chosen as 1.0 rad/sec (this corresponds to an L of 30 ft and a mean wind speed of 20 ft/sec). However, to make sure the results of the analysis were not highly sensitive to this parameter, a frequency of 0.3 rad/sec was also briefly considered. The effects of lowering ω_g from 1.0 rad/sec to 0.3 rad/sec at the nominal gains for the high M_u, low M_q case are presented in Table B-V. It may be seen that the rms gust responses are not strongly dependent on ω_g. Also, from other calculations, the slight difference beween the σ's shown in Table B-V appear to be independent of pilot gain. That is, the differences between the σ's for $\omega_g = 1.0$ rad/sec and $\omega_g = 0.3$ rad/sec remain roughly constant despite pilot gain variations from the nominals.

TABLE B-V

EFFECTS OF GUST BREAK FREQUENCY

	ω_g = 1.0 rad/sec	ω_g = 0.3 rad/sec
σ_x/σ_{u_g} (sec)	2.0	2.9
$\sigma_\theta/\sigma_{u_g}$ (deg/ft/sec)	0.81	0.68
$M_{\delta_B}\sigma_{\delta_B}/\sigma_{u_g}$ (deg/ft/sec) ...	5.9	4.7

RMS Gust Responses

The determination of the rms x, θ, and $M_{\delta_B}\delta_B$ gust responses is outlined below. The x-response provides a measure of the pilot's success in the principal task of hovering over a given point. The θ-response describes the attitude deviations of the vehicle; if the pitching variations are large, the pilot will not like the aircraft even if he is able to satisfactorily maintain position. Finally, $M_{\delta_B}\sigma_{\delta_B}$ provides a measure of both the required control power and the control effort exercised by the pilot.

The mean-squared values of x, θ, and $M_{\delta_B}\delta_B$ were calculated from the response spectra

$$\Phi_\lambda = \left|\frac{\lambda}{u_g}(j\omega)\right|^2 \Phi_{u_g} \quad \text{for} \quad \lambda = x,\ \theta,\ \text{or}\ M_{\delta_B}\delta_B \qquad \text{(B-3)}$$

φ_{u_g} is the gust spectrum of Eq. B-2 and $\lambda/u_g(j\omega)$ is the transfer function of λ to u-gust inputs with both the θ- and x-loops closed. Equation B-3 can also be put into the form

$$\varphi_\lambda = |Y_\lambda(j\omega)| \tag{B-4}$$

where $\quad Y_\lambda(j\omega) = \frac{\lambda}{u_g}(j\omega) Y_f(j\omega)$

and $\quad Y_f$ is given by Eq. B-1

The mean-squared values were computed by numerically integrating the spectra from Eq. B-4 by the method of Appendix E of Ref. 46.

Choice of Pilot Gains

The rms values of x, θ, and $M_{\delta_B}\delta_B$ were calculated for variations in pilot x- and θ-loop gains from the nominal values shown in Figs. B-2 and B-3 and Tables B-III and B-IV. The results for the two high M_q cases are given in Figs. B-4a and B-4b; the trends in these figures also apply to the low M_q cases. In the regions of interest, increasing pilot x-loop gain increases σ_θ and $M_{\delta_B}\sigma_{\delta_B}$, while σ_x decreases to a minimum value, then, while not shown, increases as instability is approached at high K_x. As the x-loop gain approaches zero, σ_x tends to infinity, and σ_θ and $M_{\delta_B}\sigma_{\delta_B}$ tend to minimum values.

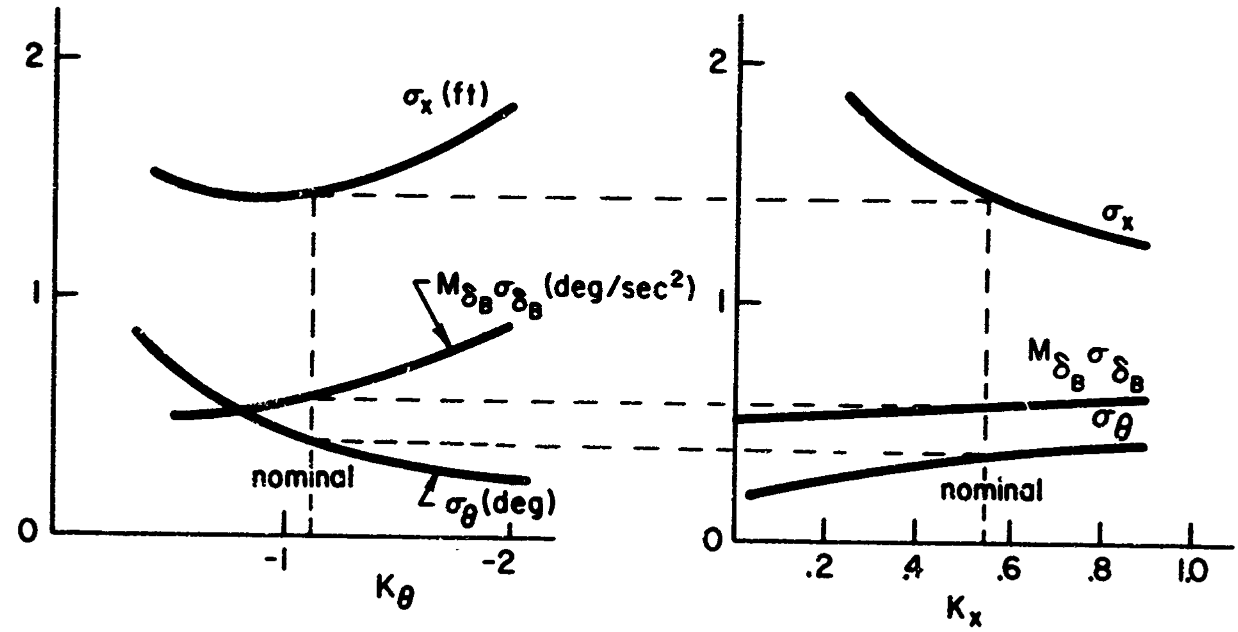
In the gain regions of interest, as pilot θ-loop gain increases

a. σ_x tends to increase, although for low M_u there is an initial region where it first decreases

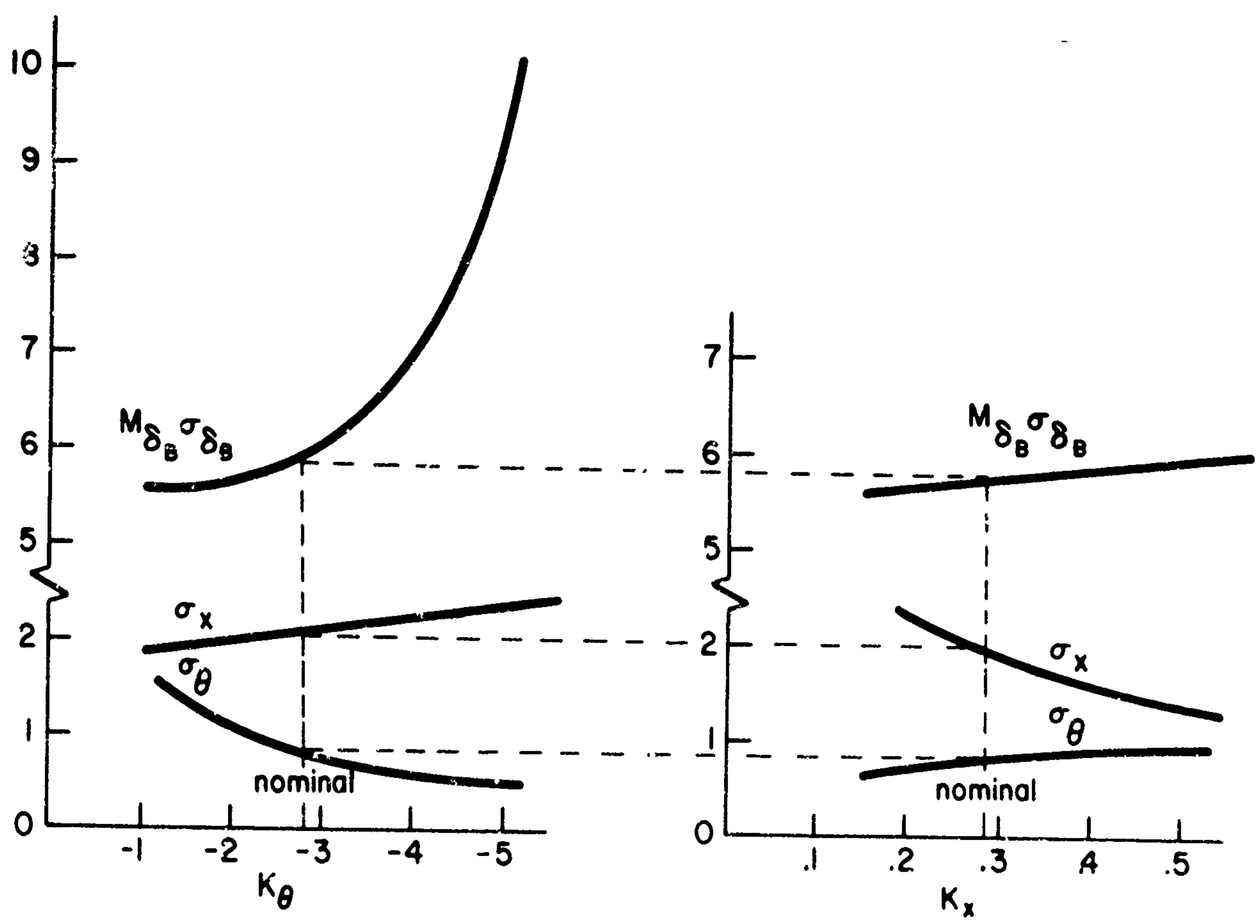
b. σ_θ decreases

c. $M_{\delta_B}\sigma_{\delta_B}$ increases

It can be seen in Figs. B-4a and B-4b that the nominal gains provide a reasonable compromise among the rms values of x, θ, and $M_{\delta_B}\delta_B$. The θ-gain gives a near-minimum θ-response without greatly increasing x or $M_{\delta_B}\delta_B$. The x-gain gives near-minimum x-response with only modest increases in θ and $M_{\delta_B}\delta_B$.



(a) Low M_u , High M_q Case



(b) High M_u , High M_q Case

Figure B-4. Variations of Gust Responses with Pilot Gains

Effects of Stability Derivatives

Having confirmed the validity of the nominal pilot gains, the effects of the rms responses of changes in the stability derivatives can now be assessed. However, because of the approximate nature of the nominal pilot parameters, small percentage changes in the gust responses should be ignored. Only gross variations can reliably be attributed to the effects of the stability derivatives.

The rms values for $\sigma_{u_g} = 5$ ft/sec for the four combinations of M_u and M_q are summarized in Table B-VI. The values all appear quite acceptable to a pilot except pitch attitude and control power variations for the high M_u cases. An attempt was made to compare these with results of an early

TABLE B-VI

RMS GUST RESPONSES

	σ_x (ft)	σ_θ (deg)	$M_{\delta_e}\sigma_{\delta_e}$ (deg/sec^2)
Low M_u, Low M_q	9	2.0	3.2
High M_u, Low M_q	9	7.0	44
Low M_u, High M_q	7	1.4	3.0
High M_u, High M_q	10	4.0	29

$\sigma_{u_g} = 5$ ft/sec

exploratory experiment on the STI fixed base simulator. Unfortunately, a direct comparison could not be made because of differences in some parameters and the limited nature of the tests. For one thing, the random-appearing sum of sinusoids used in the simulator tests was not comparable in spectral characteristics to the assumed spectrum used in the calculations. Also, the test program was small with only one pilot participating and then with only a minimum of training or learning time and without benefit of any actual helicopter experience. Figure B-5 compares time

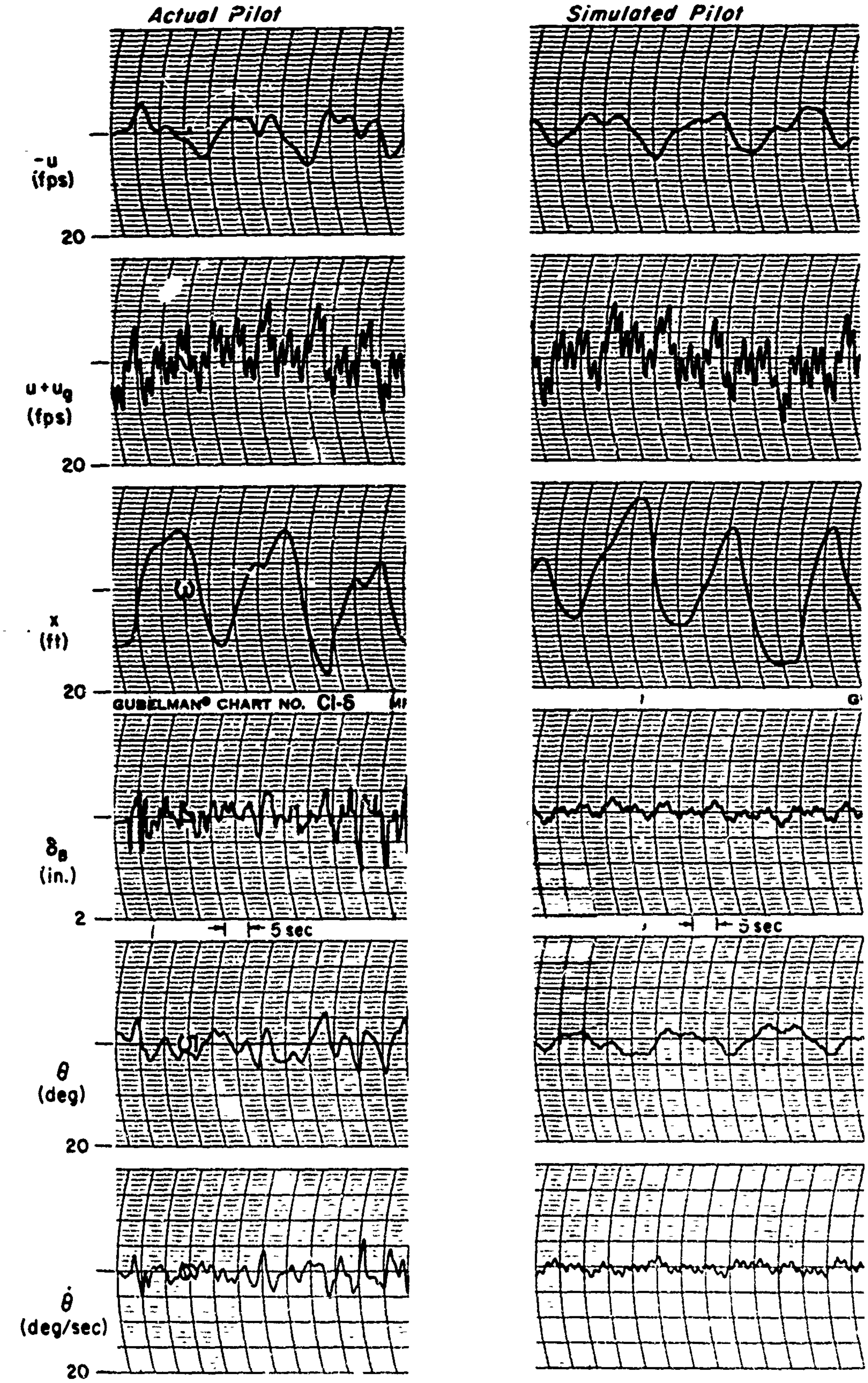


Figure B-5. Time Histories from Simulated and Actual Pilot Runs for Low M_u, High M_q Case in Hover Fixed Base Simulator

histories of this pilot and the "optimum" pilot of Tables B-III and B-IV for the low M_u, high M_q case. Based upon a σ_{u_g} of 5 ft/sec the actual pilot of Fig. B-5 has the following responses:

$$\sigma_\theta = 3.6 \text{ deg}$$

$$\sigma_x = 10.1 \text{ ft}$$

$$M_{\delta_B}\sigma_{\delta_B} = 7.2 \text{ deg/sec}^2$$

Compared to the analytical pilot of Table VI

$$\sigma_\theta = 1.4 \text{ deg}$$

$$\sigma_x = 7.0 \text{ ft}$$

$$M_{\delta_B}\sigma_{\delta_B} = 3.0 \text{ deg/sec}^2$$

The biggest discrepancies appear in the control responses where those of the actual pilot are somewhat nonlinear. It is not known at this time whether training could reduce this remnant or whether the remnant is largely a result of the nature of the task. This is an area which requires further work and should include experimental measurements of the pilot's remnant.

To provide additional insight into the numerical results an attempt was made to obtain approximate literal expressions for the response ratios for $M_{\delta_B}\sigma_{\delta_B}/\sigma_{u_g}$, $\sigma_\theta/\sigma_{u_g}$, and σ_x/σ_{u_g}. One approximation does predict the $M_\delta\sigma_\delta$ results of the four extreme cases of Table B-VI with an average error of about 20 percent and is given by

$$\left(\frac{M_\delta \sigma_\delta}{\sigma_{u_g}}\right)^2 \doteq M_u^2 \left[\frac{1 + \dfrac{2\zeta_p''\omega_p''}{\omega_{u_g}} + (T_{L_\theta}')^2(\omega_p'')^2}{2\zeta_p'' \dfrac{\omega_{u_g}}{\omega_p''} + 2\zeta_p'' + \dfrac{\omega_p''}{\omega_{u_g}}}\right] \left(\frac{\text{rad/sec}^2}{\text{ft/sec}}\right)^2 \qquad \text{(B-5)}$$

The bracketed term in Eq. B-5 was evaluated for several cases including variations in closure gain as well as the stability derivative. The resulting values were essentially constant, leading to the linear relationship given by

$$\frac{M_{\delta}\sigma_{\delta}}{\sigma_{u_g}} \doteq [1.4] M_u \quad \frac{\text{rad/sec}^2}{\text{ft/sec}}$$

$$\text{or} \qquad \doteq [80] M_u \quad \frac{\text{deg/sec}^2}{\text{ft/sec}} \tag{B-6}$$

where the mean constant in the bracket was obtained by dividing M_u into the actual $M_{\delta_B}\sigma_{\delta_B}/\sigma_{u_g}$ and not the approximate expression of Eq. B-5.

A fairly accurate approximation was developed for the mean square variation in position (see Ref. 45)

$$\left(\frac{\sigma_x}{\sigma_{u_g}}\right)^2 \doteq \frac{\omega_x''}{2\zeta_x''\omega_g} \frac{1}{(K_x)^2} \tag{B-7}$$

An attempt to obtain a simple approximation for $\sigma_\theta/\sigma_{u_g}$ was unsuccessful.

The effects of X_u on σ_x for the two high M_q cases and on σ_θ for the high M_u, high M_q case were evaluated by computing their partial derivatives with respect to X_u. The results appear in Table B-VII. None of the changes are very important, considering the extreme change in X_u that was used.

The gust responses are all for $X_{\delta_B} = 0$. The rms values of x, θ, and $M_{\delta_B}\delta_B$ were also calculated for the high M_u, low M_q case with $X_{\delta_B}/M_{\delta_B} = -5$ ft. The results were quite close to the rms values for $X_{\delta_B} = 0$; the maximum change was 10 percent. Thus, realistic values of X_{δ_B} appear to have little effect on the gust responses.

TABLE B-VII

X_u EFFECTS ON GUST RESPONSES

	$\dfrac{\partial\left(\frac{\sigma_x}{\sigma_{u_g}}\right)}{\partial(X_u)}$	$\Delta(X_u)$	$\Delta\sigma_x$ * (ft)
Low M_u, High M_q	−1.53	−0.3	2.3
+High M_u, High M_q	−0.292	−0.3	0.44

	$\dfrac{\partial\left(\frac{\sigma_\theta}{\sigma_{u_g}}\right)}{\partial(X_u)}$	$\Delta(X_u)$	$\Delta\sigma_\theta$ * (deg)
+High M_u, High M_q	1.09	−0.3	−1.6

*These changes are for $\sigma_{u_g} = 5$ ft/sec

+These were calculated at nominal K_x, but at $\kappa_\theta = 1.44\ \text{sec}^{-1}$

II. LATERAL-DIRECTIONAL CONTROL DURING APPROACH

Configuration Review

Four NASA and two NRCC configurations of Refs. 15 and 16 respectively were selected as being representative of most all situations in the two experiments. The derivatives (obtained from the referenced reports) and the computed transfer function factors are given in Tables B-VIII and B-IX, respectively. Only the NRCC configurations with the lowest value of N_v could be considered on a par with the NASA tests due to the gust effects, as explained in Section III-B. For purposes of comparison herein, it is assumed that the control sensitivity is always at its optimum value. That is, we are interested in the best ratings for a given set of dynamics.

In general, the two model simulators were representative of their respective classes, i.e., the tandem rotor configurations were low in roll damping (L_p) and directional stability (N_v) as opposed to the single rotor cases. Only in a few situations did the NASA model approach a flight condition similar to the NRCC model. NASA's Condition E* (with $L_v = 0$) and the NRCC Condition 2 is one such overlap and the pilot ratings were approximately the same. The lateral transfer functions are based on the following equations of motion (the product of inertia effects are considered negligible):

$$\begin{aligned}
(s - Y_v)v \quad & - g\varphi & + U_o r &= Y_{\delta_A}\delta_A \\
-L_v v \quad & + s(s - L_p)\varphi & &= L_{\delta_A}\delta_A \\
-N_v v \quad & & + (s - N_r)r &= N_{\delta_r}\delta_r
\end{aligned} \tag{B-8}$$

The coupled moment derivatives (L_r and N_p) and the cross-control derivatives (L_{δ_r} and N_{δ_A}) were said to be negligible.

*Actually, the pilot rating for this situation ($L_v = 0$) was not given in Ref. 15, but was obtained by telephone conversation with the author.

TABLE B-VIII

SUMMARY OF DERIVATIVES FOR NASA AND NRCC FLIGHT CONDITIONS

DIMENSIONAL DERIVATIVE	CONFIGURATION					
	NASA 1(A)	NASA 4(D)	NASA 11(K)	NASA 16(E*)	NRCC 1	NRCC 2
Y_v	$\doteq 0$	$\doteq 0$	$\doteq 0$	$\doteq 0$	$\doteq 0$	$\doteq 0$
Y_{δ_A}	1.3	1.3	1.3	1.3	1.0	1.0
L_v (L_β)	−0.014 (−1.06)	−0.014 (−1.06)	−0.014 (−1.06)	0 (0)	0 (0)	0 (0)
L_p	−1.5	−1.5	−1.5	−1.5	−4.2	−4.2
L_{δ_A}	0.40	0.40	0.40	0.40	0.41	0.41
N_v (N_β)	0.0013 (0.10)	0.0013 (0.10)	0.013 (1.00)	0.0053 (0.4)	0.01 (0.506)	0.01 (0.506)
N_r	−0.2	−1.0	−2.0	−1.0	−1.0	−0.1
N_{δ_r}	0.2	0.2	0.2	0.2	0.5	0.5

TABLE B-IX

SUMMARY OF LATERAL TRANSFER FUNCTION FACTORS FOR NASA AND NRCC FLIGHT CONDITIONS

		NASA 1(A)	NASA 4(D)	NASA 11(K)	NASA 16(E*)	NRCC 1	NRCC 2
Δ Denominatcr	$\frac{1}{T_s}$ (ζ_{SR})	0.156	0.944	(0.98)	0	0	0
	$\frac{1}{T_R}$ (ω_{SR})	1.66	1.65	(1.62)	1.5	4.2	4.2
	ζ_d	−0.096	−0.089	0.28	0.79	0.70	0.07
	ω_d	0.59	0.54	0.59	0.63	0.711	0.711
$N_{\varphi\delta_A}$	κ	0.40	0.40	0.40	0.40	0.40	0.40
	$\zeta_\varphi \left(\frac{1}{T_{\varphi_1}}\right)$	0.26	(0.061)	(0.76)	0.79	0.70	0.07
	$\omega_\varphi \left(\frac{1}{T_{\varphi_2}}\right)$	0.30	(0.893)	(1.19)	0.63	0.711	0.711
$sN_{\psi\delta_A}$	κ	0.0017	0.0017	0.0017	0.0068	0.01	0.01
	ζ_ψ	0.24	0.24	0.24	0.24	0.58	0.58
	ω_ψ	3.15	3.15	3.15	3.15	3.63	3.63
$N_{r\delta_r}$	κ	0.2	0.2	0.2	0.2	0.5	0.5
	$\frac{1}{T_{r_1}}$	1.66	1.66	1.66	1.5	4.2	4.2
	ζ_r	−0.16	−0.16	−0.16	0	0	0
	ω_r	0.52	0.52	0.52	0	0	0

The two experiments are defined as follows:

	NASA (Ref. 15)	NRCC (Ref. 16)
Task..............	IFR and VFR 3^{o} approach	VFR 11^{o} approach
Simulator.........	Tandem rotor	Single rotor
Type..............	Vertol 107	H-13G
Approximate wt....	13,000 lb	2,900 lb
Speed, U_o.........	76 ft/sec	50.6 ft/sec
Wind..............	Wind from all directions, 5 to 15 knots with occasional gusts to 25 knots	Flight path into wind with simulated 8.9 ft/sec rms side gust

For the configuration reviews which follow, all $\varphi \rightarrow \delta_a$ closures were made with pilot lead equal to roll time constant, $T_L = T_R$, and the gain was chosen to give 30 deg phase margin with crossover frequencies greater than 2 rad/sec and gain margins of approximately 5 db. In NASA Flight Condition 4(d) it was thought that the inner loop gain might be critical in establishing the crossover frequency of the outer heading loop, but was found later to be noncritical [see discussion under Flight Condition 4(D)]. In all closures the pilot's time delay (τ) was selected at a loose 0.4 sec. None of the above items were found to be critical and were selected as a standard of comparison.

NASA FLIGHT CONDITION 1(A)*, PR = 6

$\varphi \rightarrow \delta_A$ Closure

Inner loop closure, Fig. B-6a, provides only a limited potential (low $\zeta_\varphi \omega_\varphi$) for damping the dutch roll. Damping $\zeta_d' \omega_d'$ was only increased to a positive 0.13 sec^{-1} after the first closure (the symbol ▮ locates the closed-loop poles).

$\psi \rightarrow \delta_A$ Closure, with $\varphi \rightarrow \delta_A$ Inner Loop

The poor damping of $\zeta_d' \omega_d' \doteq \zeta_\varphi \omega_\varphi$ prevents the attainment of a good crossover frequency (Fig. B-6b), which is less than 0.2 rad/sec.

*The letter in parentheses is NASA's designation.

NASA FLIGHT CONDITION 4(D), PR = 5-1/2

$\varphi \rightarrow \delta_A$ Closure

Even though the basic dutch roll mode is unstable, the pilot with low gain could obtain the characteristics he likes (see Fig. B-7a).

$\psi \rightarrow \delta_A$ Closure, with $\varphi \rightarrow \delta_A$ Inner Loop

The interesting aspect of this configuration is the poor heading closure of Fig. B-7b. For a relatively large gain margin of 9 db, the crossover frequency is a low 0.16 rad/sec at 30 deg of phase margin. The primary reason for the low ω_c is the small $1/T_{\varphi_1}$ zero of the φ/δ_a numerator. This is responsible for the pilot's comment "aircraft will not follow into desired turns using lateral control."

At first thought, it would appear that the performance of the heading control with lateral stick would depend on the pilot's gain used to close the inner loop. However, closing the inner loop at a much lower gain (increasing the damping of the closed-loop pole nearest the origin, Fig. B-7a) increased the crossover frequency (ω_c) of the heading to 0.22, but reduced the gain margin to less than 5 db for the same 30 deg phase margin. Therefore, heading control is little changed by the pilot's selection of inner loop gain (this is consistent with the findings of Ref. 17).

NASA FLIGHT CONDITION 11(K), PR = 2-1/2

$\varphi \rightarrow \delta_A$ Closure

Adequate damping is easily achieved in the roll closure due to the large $1/T_{\varphi_1}$ of 0.76 (see Fig. B-8a).

$\psi \rightarrow \delta_A$ Closure, with $\varphi \rightarrow \delta_A$ Inner Loop

The large $\zeta_d'\omega_d'$ (due to the large $1/T_{\varphi_1}$), as shown in Fig. B-8b, provides sufficient bandwidth to obtain a faster heading response than for most other configurations, which correlates well with the pilot's comment "Good turning response to bank."

NASA FLIGHT CONDITION 16(E*)**, PR = 3

$\varphi \rightarrow \delta_A$ Closure

Because of the ω_φ, ω_d cancellation (Fig. B-9a), the dutch roll cannot be damped with δ_a. In this case, the basic damping of this mode is acceptable for a good rating.

$\psi \rightarrow \delta_A$ Closure, with $\varphi \rightarrow \delta_A$ Inner Loop

For the reason mentioned under the $\varphi \rightarrow \delta_A$ closure, the value of $\zeta_\varphi \omega_\varphi$ ($= \zeta_d \omega_d$) is adequate to also obtain an acceptable rating for the heading control. This outer loop closure, Fig. B-9b, has a crossover frequency of $\omega_c = 0.3$.

NRCC FLIGHT CONDITION 1, $PR_{min} = 6$

$\varphi \rightarrow \delta_A$ Closure

The condition is typical of the configurations where the dutch roll is poorly damped and the lateral stick cannot damp this mode. The closure of Fig. B-10a looks the same as that given in Fig. B-9a because of the ω_φ, ω_d cancellation.

$\psi \rightarrow \delta_A$ Closure, with $\varphi \rightarrow \delta_A$ Inner Loop

As shown in Fig. B-10b, the dutch roll goes unstable for very low gain. The gain margin limits the crossover frequency to less than 0.1 rad/sec in order to achieve a 5 db gain margin in a good K/s region.

---

*Refers to NASA Condition E with $L_v = 0$.

**This condition was selected because it is similar to NRCC Flight Condition 1. The open-loop transfer functions, Table B-IX, are essentially the same except for the roll subsidence ($\doteq L_p$), which is approximately one-third that used by NRCC. As it turns out, the closed-loop aspects are identical even though the pilot would possibly be required to use 0.67 sec lead in the $\varphi \rightarrow \delta_a$ closure as opposed to 0-0.25 sec lead in the NRCC condition. The difference in these lead requirements is a difference in rating of less than one-half a Cooper point (see Fig. 3). Although the ratings were the same, the difference of one rating point is within the error of the Cooper rating system. This is an interesting comparison since the two helicopters are vastly different in size and weight.

NRCC* FLIGHT CONDITION 2, $PR_{min} = 3$

$\varphi \rightarrow \delta_A$ Closure

The closure of Fig. B-11a is identical to NASA Condition 16(E*) of Fig. B-9a when the pilot selects his lead equal to the roll subsidence ($1/T_R \doteq 1/T_L$).

$\psi \rightarrow \delta_A$ Closure, with $\varphi \rightarrow \delta_A$ Inner Loop

The crossover frequency of the heading control $\psi \rightarrow \delta_A$ is an ample 0.4 rad/sec (Fig. B-11b). The improvement (higher crossover frequency) over the similar NASA Condition 16(E*) is due to the higher roll subsidence of this condition.

*The NRCC Flight Conditions 1 and 2, as defined in Tables B-VIII and B-IX were selected as being comparable to NASA conditions where L_v is zero and N_v small. The NRCC ratings were taken as the best attainable at the optimum gain. Only ratings from the circuit approach task (as opposed to the hover task) were considered comparable.

Helicopter: $\frac{\phi}{\delta_a} = \frac{N_{\phi/\delta_a}}{\Delta} = \frac{.4\left[s^2 + 2(.25)(.30)s + (.30)^2\right]}{(s+.156)(s+1.66)\left[s^2 + 2(-.096)(.59)s + (.59)^2\right]}$

Pilot: $Y_{p_\phi} = K_{p_\phi} e^{-.4s}(s+1.66)$

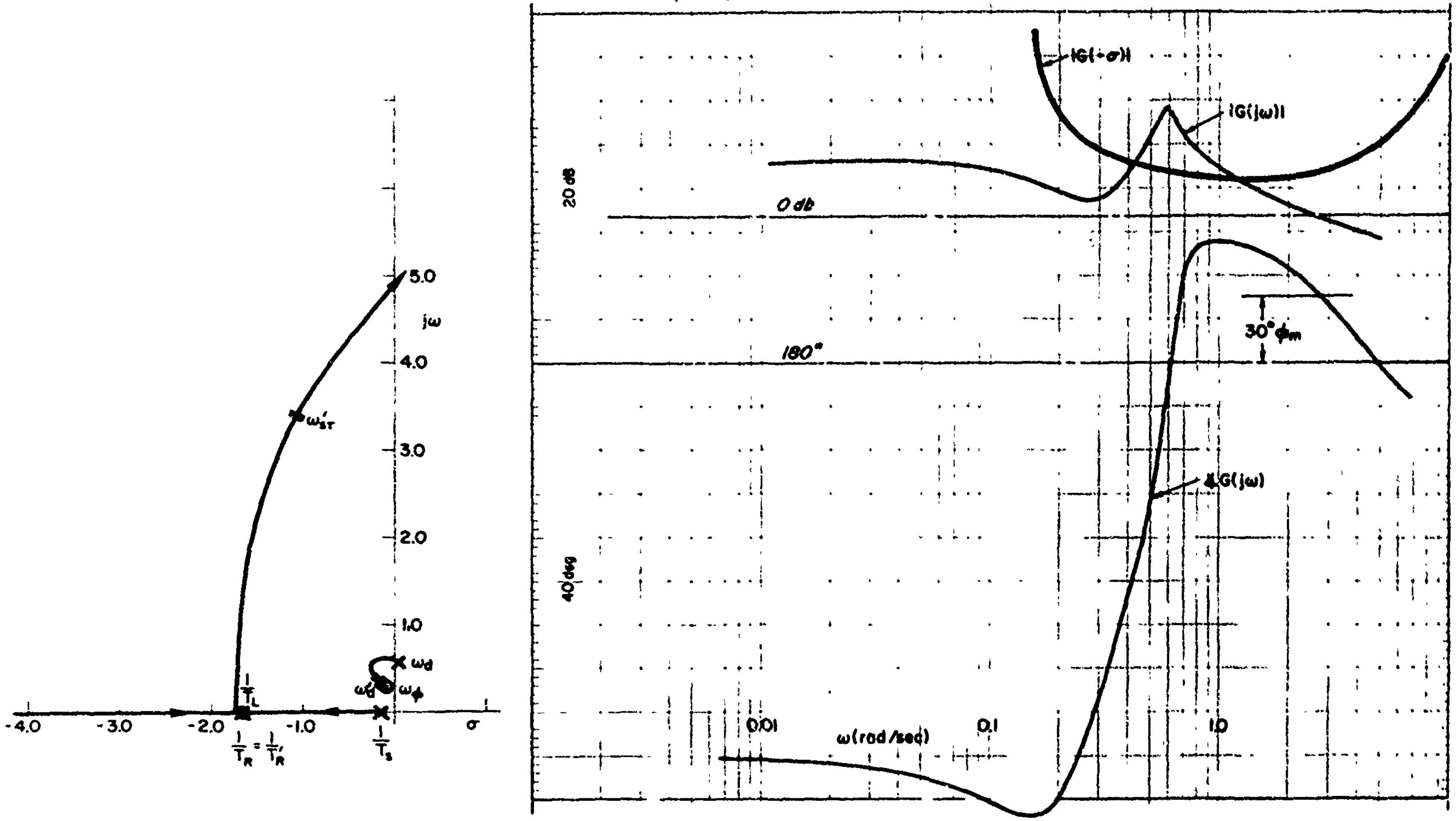


Figure B-6a. NASA Flight Condition 1(A), Roll Closure

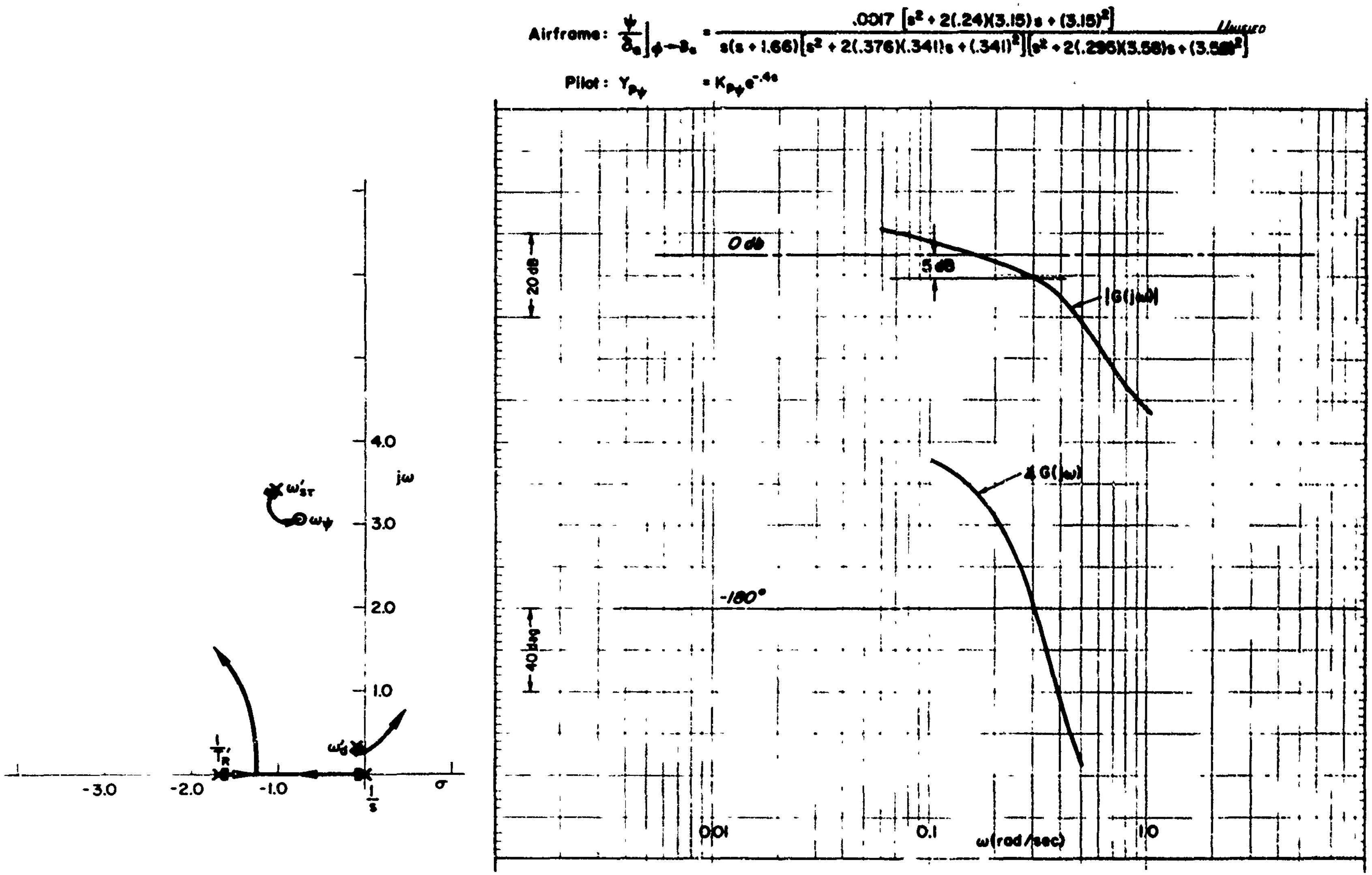


Figure B-6b. NASA Flight Condition 1(A), Heading Closure

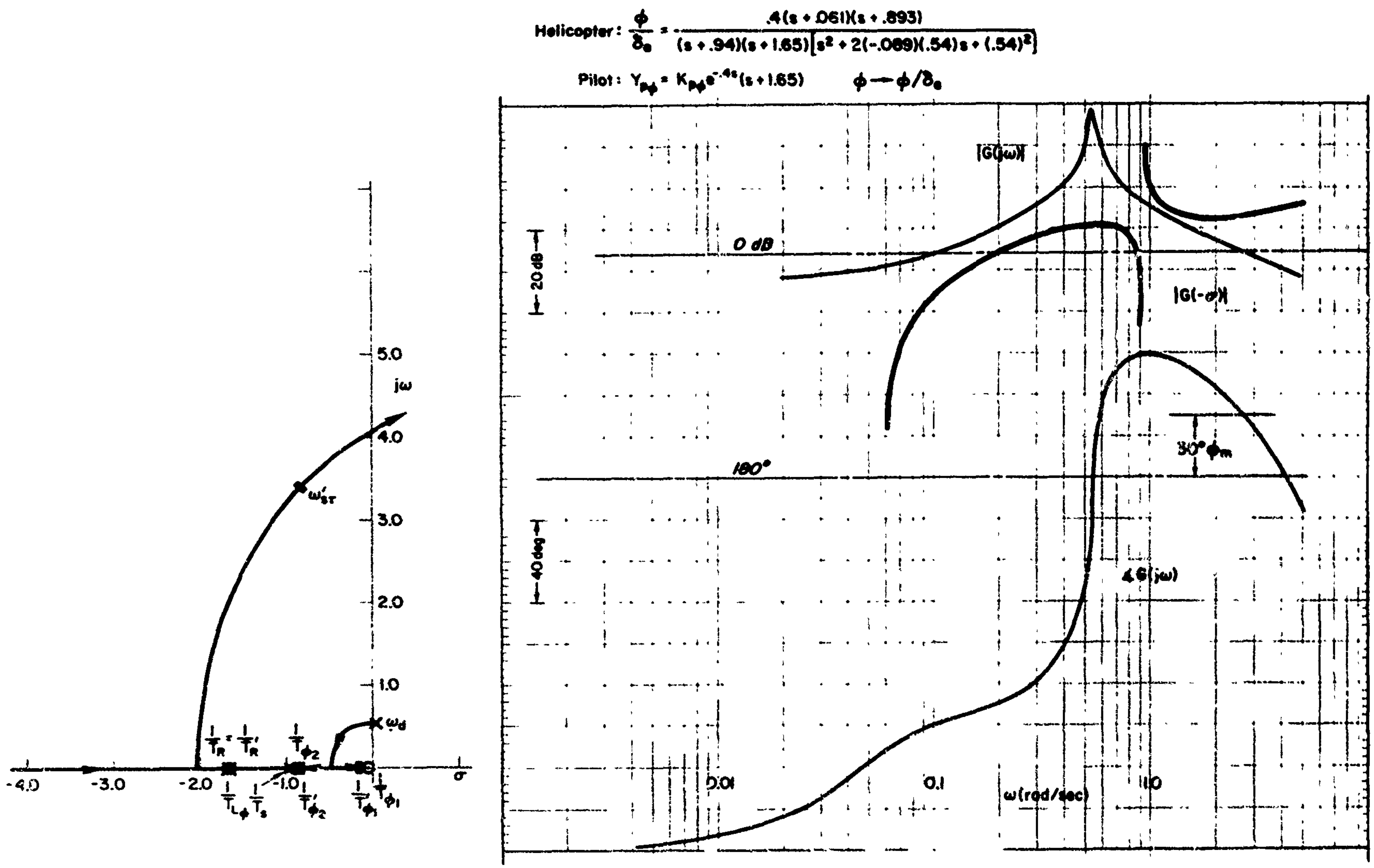


Figure B-7a. NASA Flight Condition 4(D), Roll Closure

Airframe: $\left.\frac{\psi}{\delta_a}\right]_{\phi \to \delta_a} = \frac{.0017[s^2 + 2(.24)(3.15)s + (3.15)^2]}{s(s+.197)(s+.86)(s+1.65)[s^2 + 2(.238)(3.42)s + (3.42)^2]}$

Pilot: $Y_{p_\psi} = K_{p_\psi} e^{-.4s}$

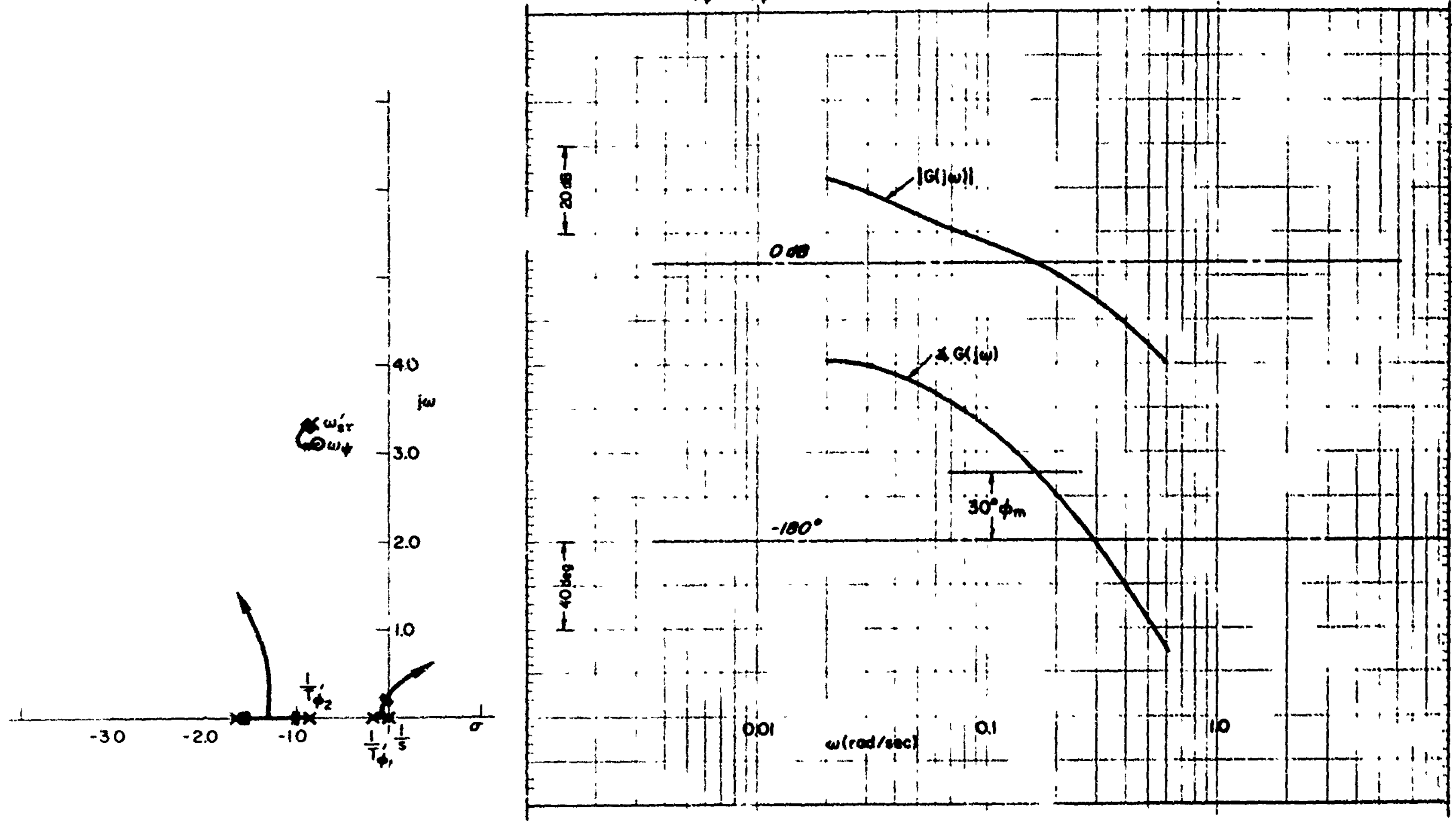


Figure B-7b. NASA Flight Condition 4(D), Heading Closure

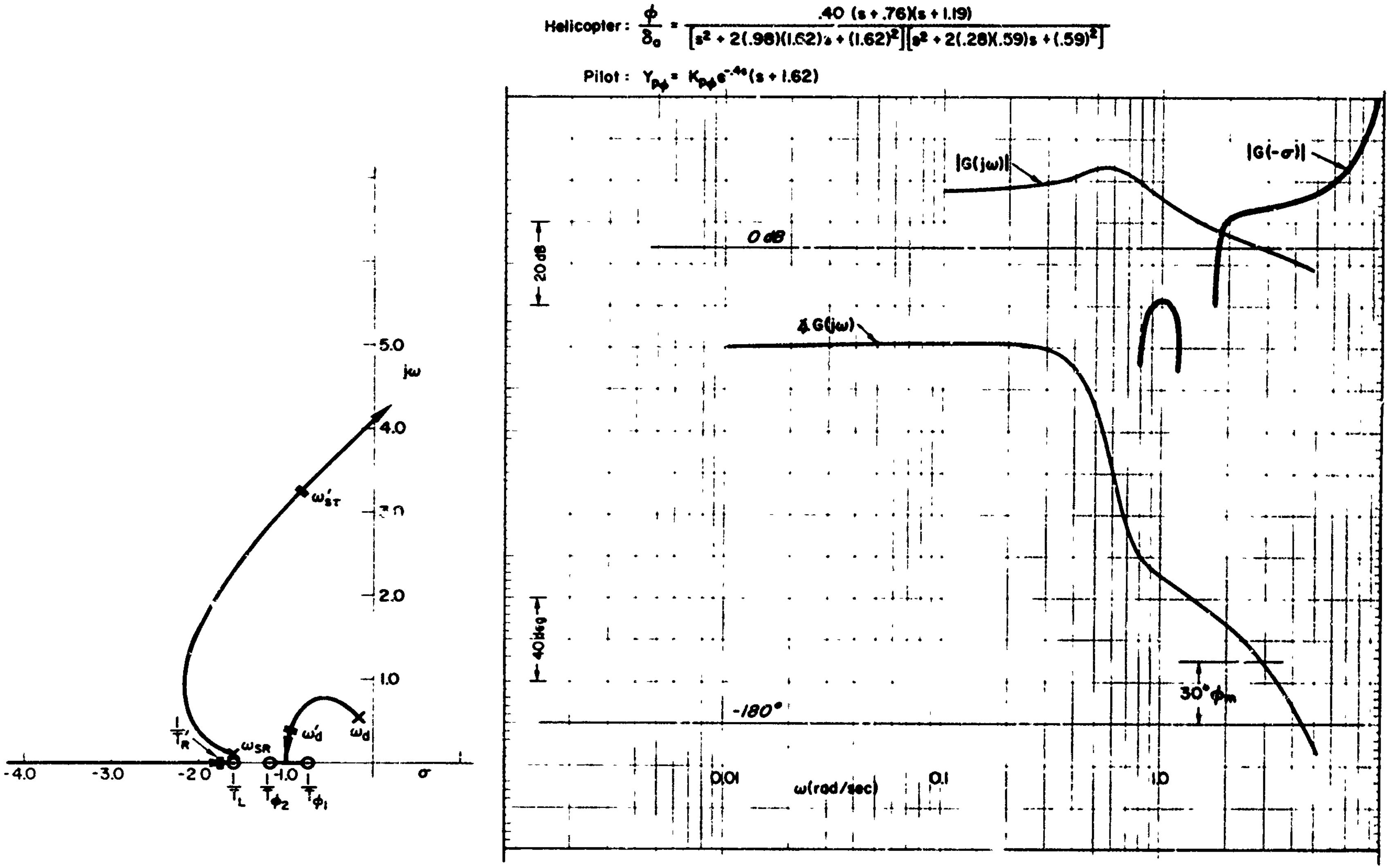


Figure B-8a. NASA Flight Condition 11(K), Roll Closure

$$\frac{\psi}{\delta_a}\bigg|_{\phi \to \delta_a} = \frac{.0017\left[s^2 + 2(.24)(3.15)s + (3.15)^2\right]}{s(s+1.73)\left[s^2 + 2(.911)(1.04)s + (1.04)^2\right]\left[s^2 + 2(.267)(3.40)s + (3.40)^2\right]}$$

$$Y_{p\psi} = K_{p\psi} e^{-.4s}$$

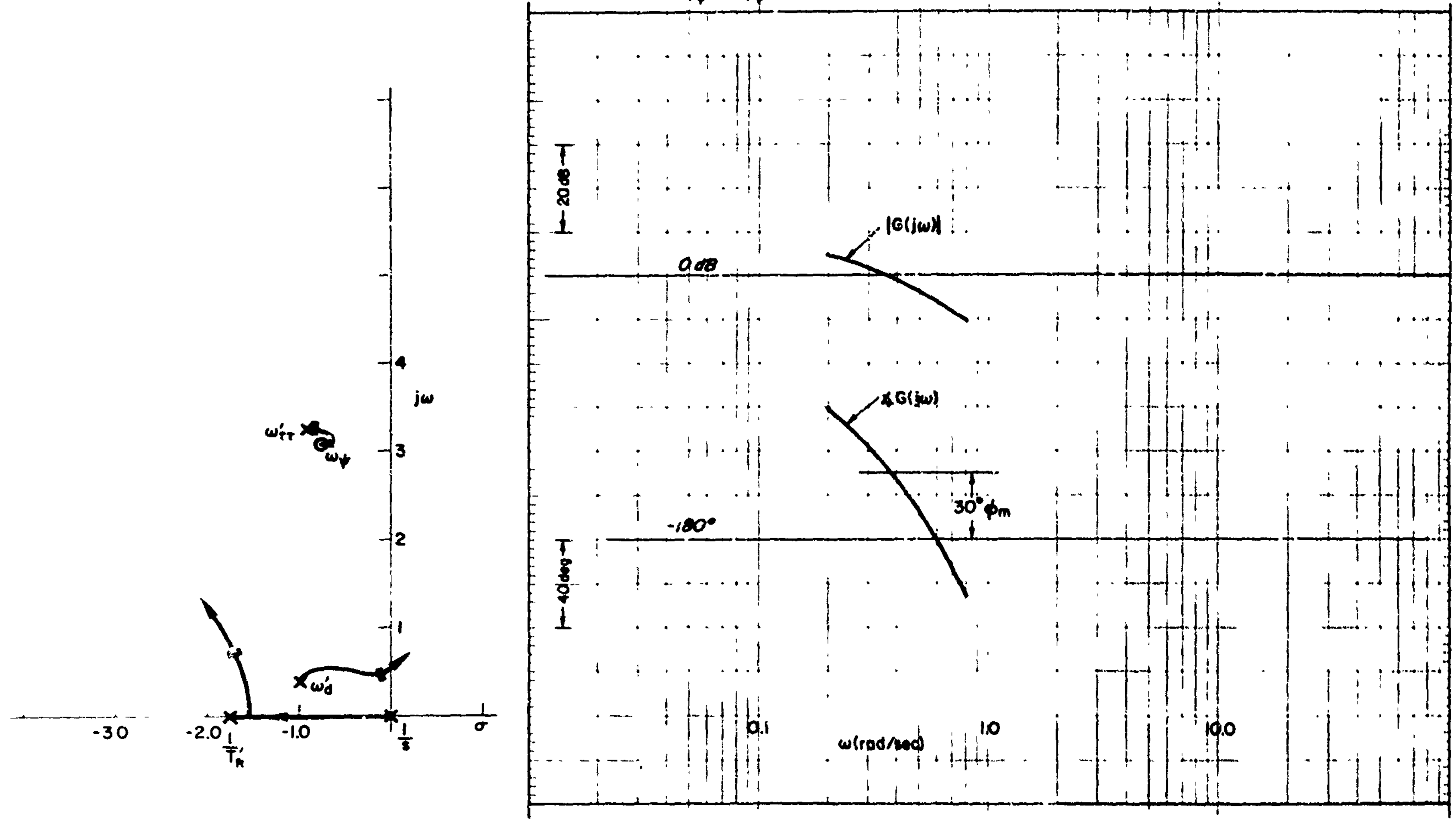


Figure B-8b. NASA Flight Condition 11(K), Heading Closure

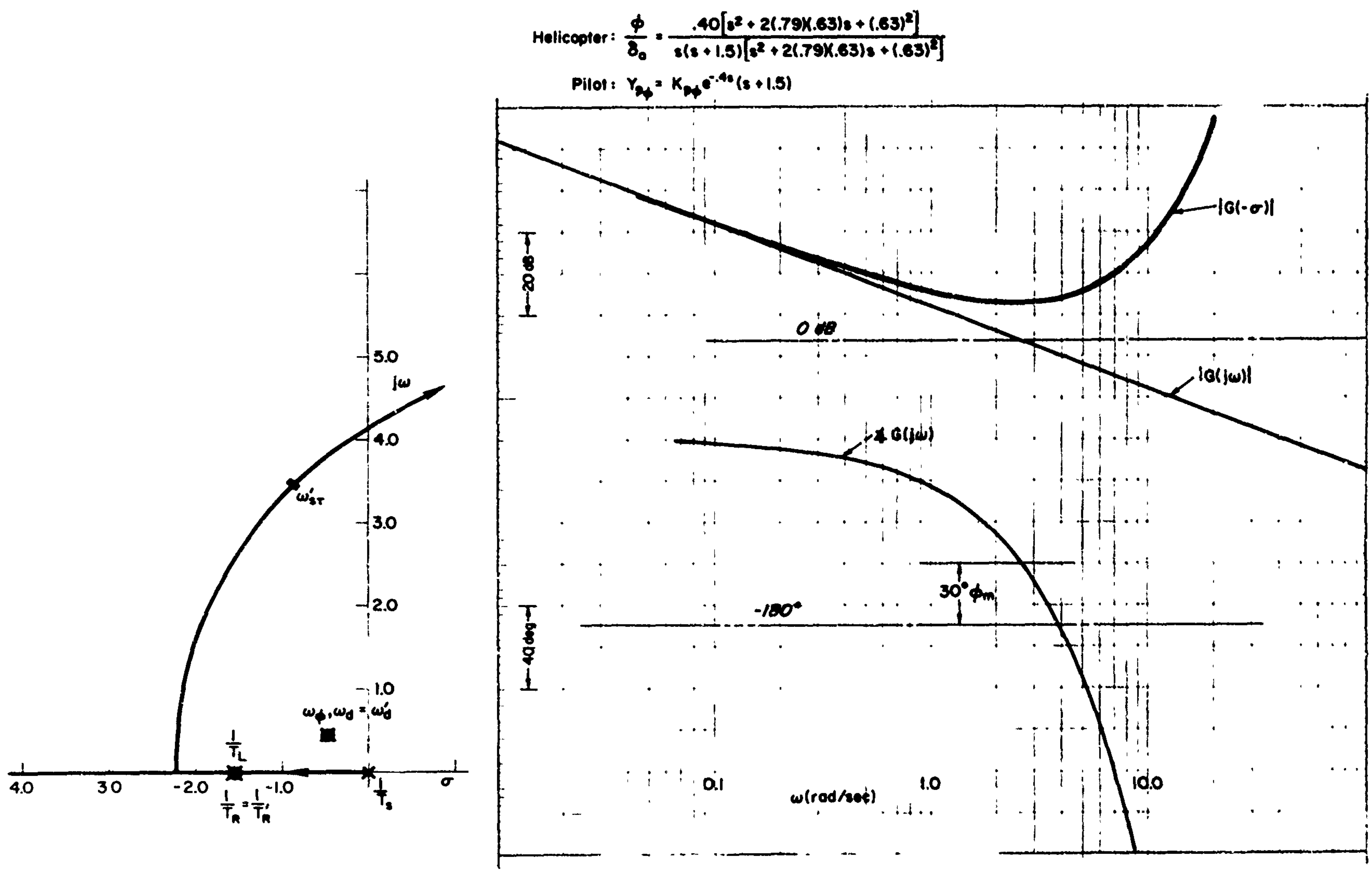


Figure B-9a. NASA Flight Condition 16(E*), Roll Closure

$$\left.\frac{\psi}{\delta_a}\right]_{\phi \to \delta_a} = \frac{.0068\left[s^2 + 2(.24)(3.15)s + (3.15)^2\right]}{s(s+1.5)\left[s^2 + 2(.79)(.63)s + (.63)^2\right]\left[s^2 + 2(.24)(3.40)s + (3.40)^2\right]}$$

$$Y_{p\psi} = K_{p\psi} e^{-.4s}$$

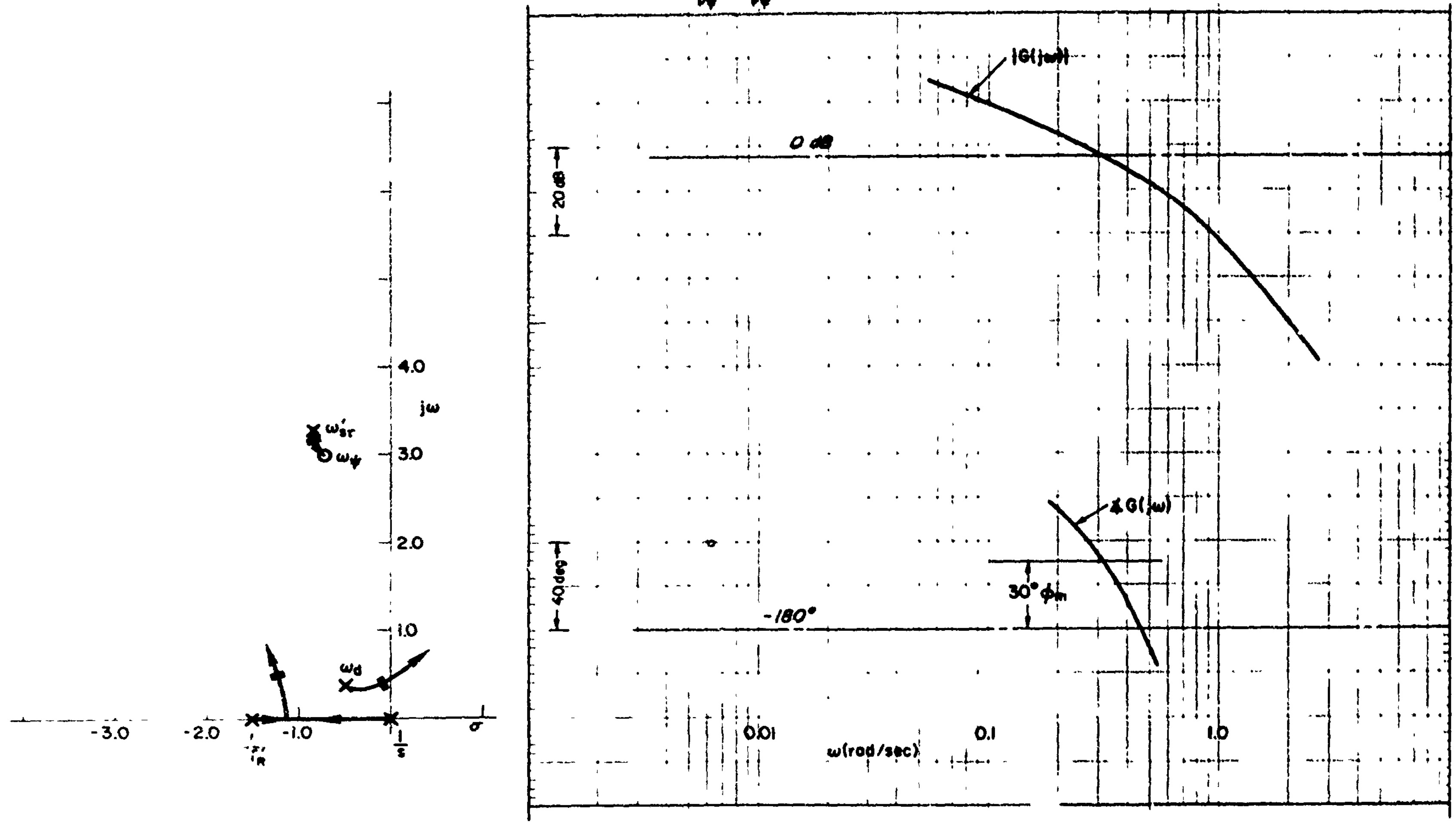


Figure B-9b. NASA Flight Condition 16(E*), Heading Closure

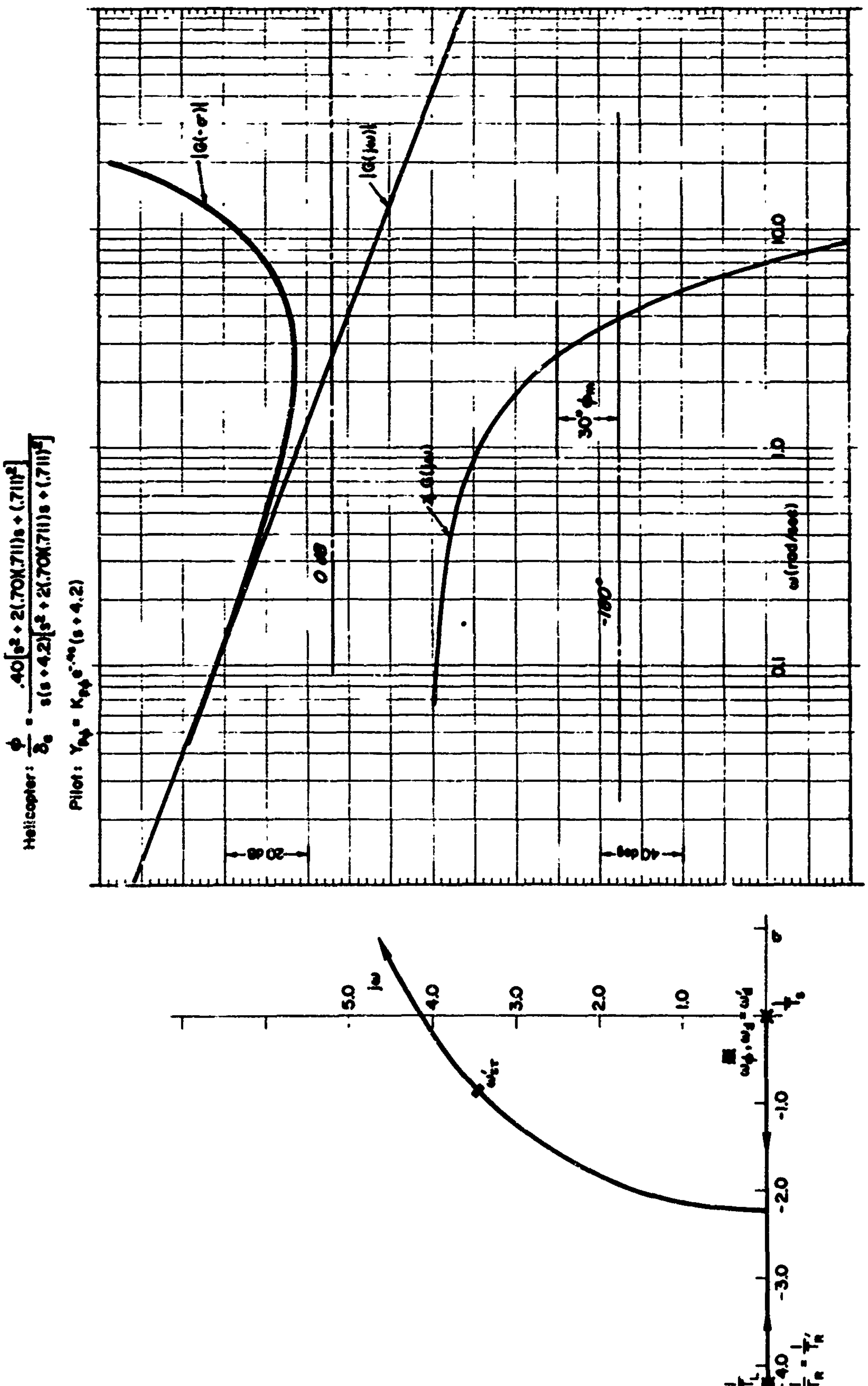


Figure B-10a. NRCC Flight Condition 1, Roll Closure

$$\left.\frac{\psi}{\delta_a}\right|_{\phi\rightarrow\delta_a} = \frac{.01\left[s^2 + 2(.58)(3.63)s + (3.63)^2\right]}{s(s+4.2)\left[s^2 + 2(.7)(.71)s + (.71)^2\right]\left[s^2 + 2(.23)(3.56)s + (3.56)^2\right]}$$

$$Y_{p\psi} = K_{p\psi}e^{-.4s}$$

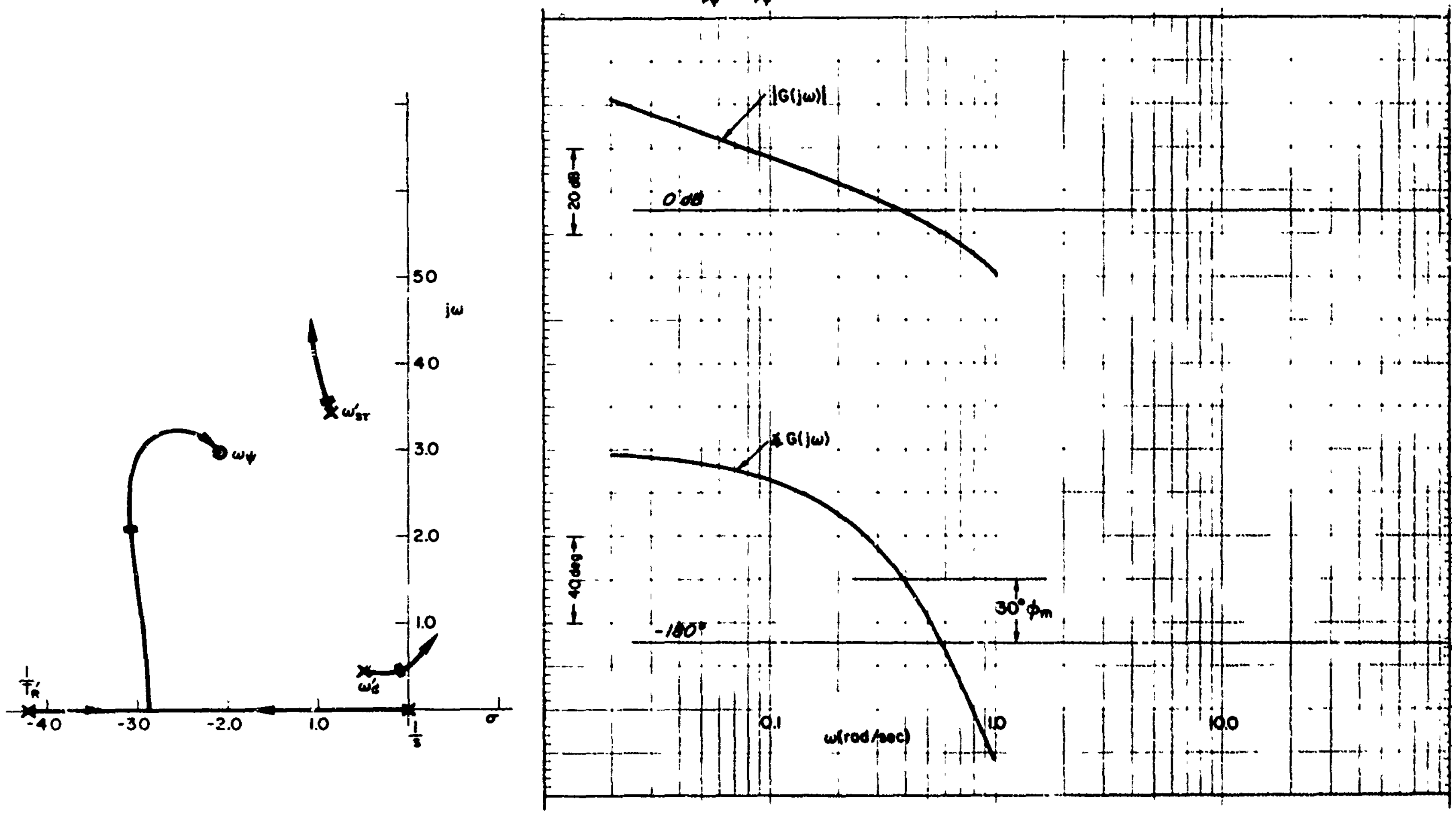


Figure B-10b. NRCC Flight Condition 1, Heading Closure

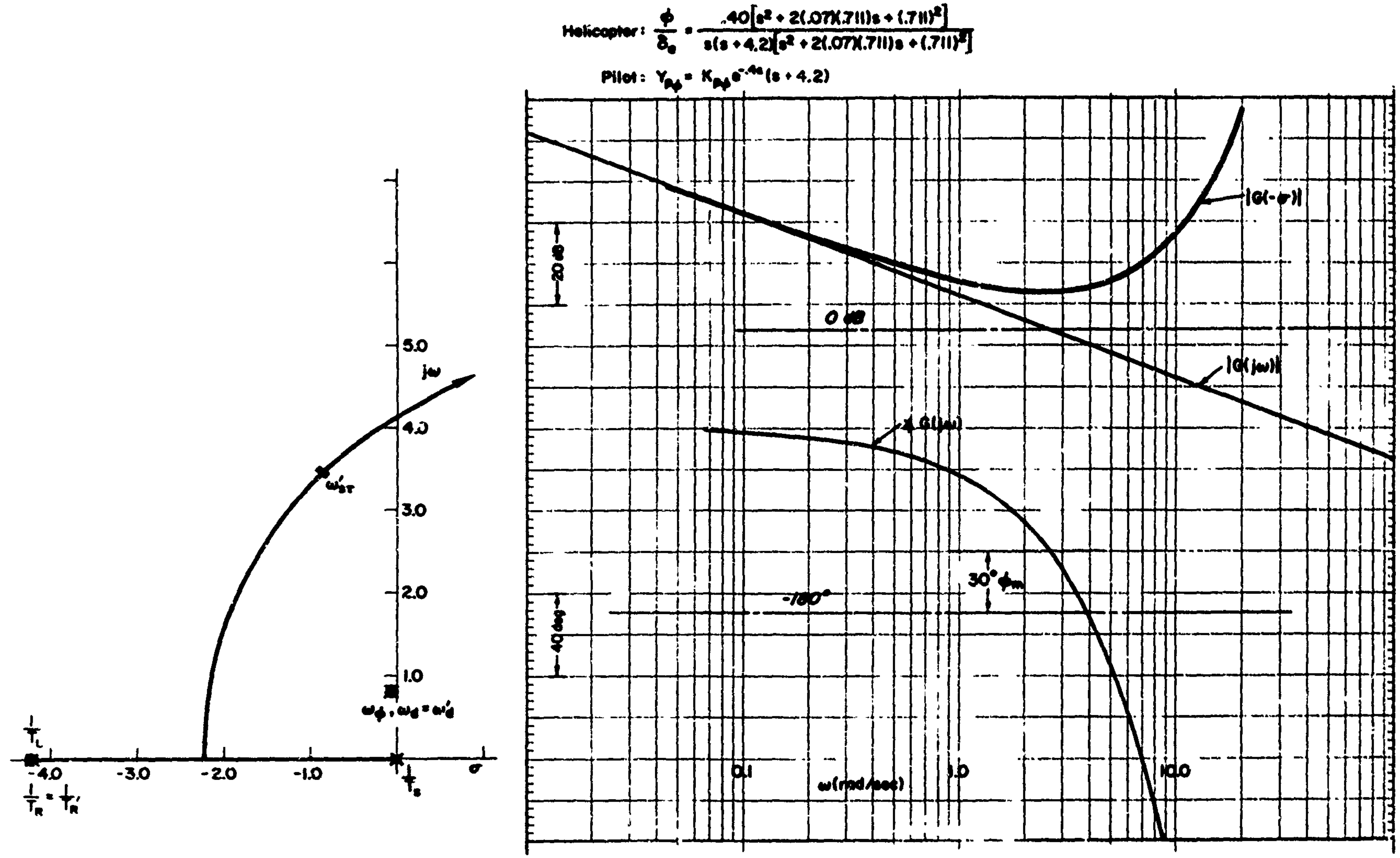


Figure B-11a. NRCC Flight Condition 2, Roll Closure

$$\left.\frac{\psi}{\delta_a}\right]_{\phi \to \delta_a} = \frac{.01\left[s^2 + 2(.58)(3.63)s + (3.63)^2\right]}{s(s+4.2)\left[s^2 + 2(.07)(.711)s + (.711)^2\right]\left[s^2 + 2(.23)(3.56)s + (3.56)^2\right]}$$

$$Y_{p_\psi} = K_{p_\psi} e^{-.4s}$$

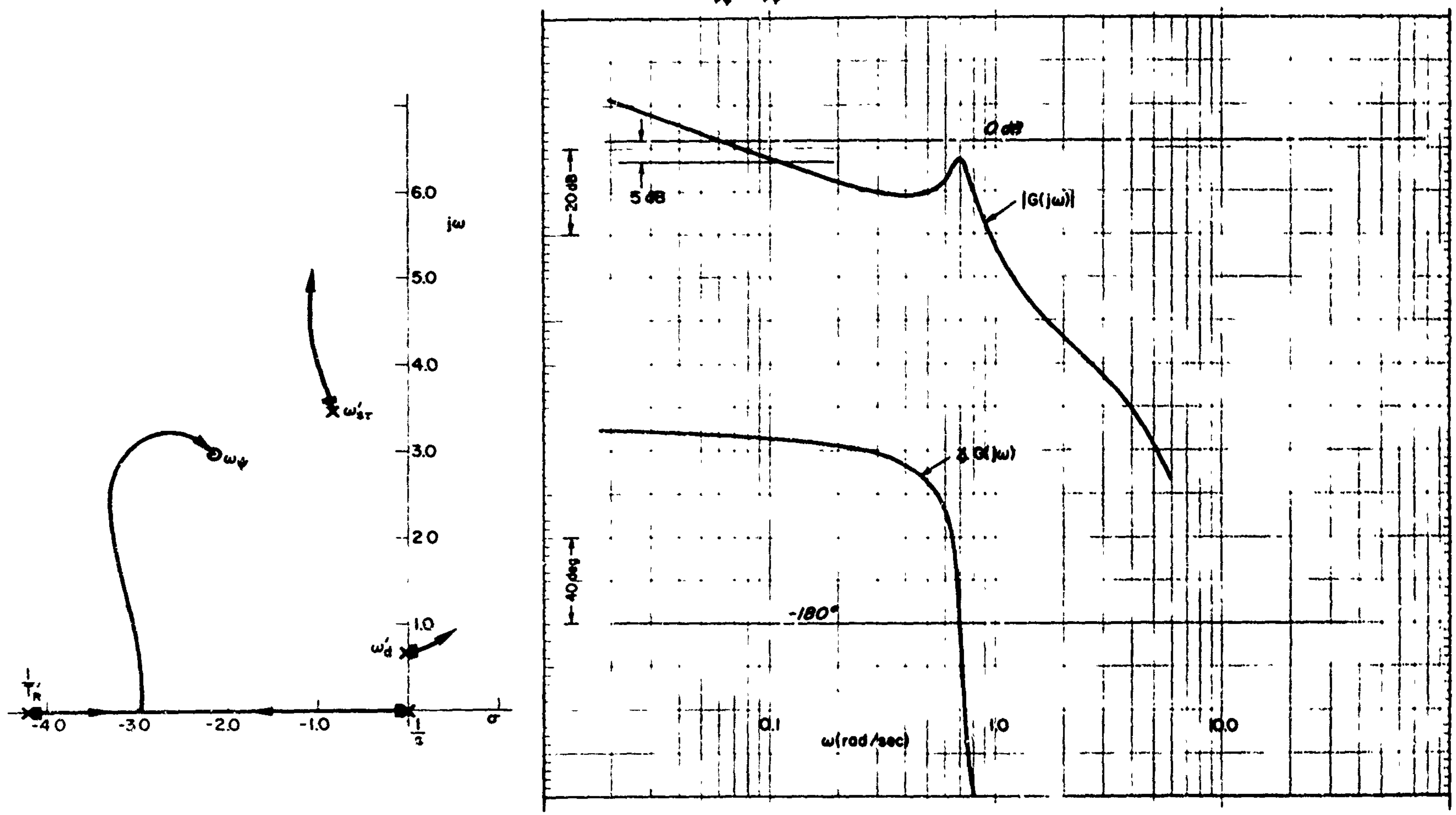


Figure B-11b. NRCC Flight Condition 2, Heading Closure

III. DIRECTIONAL APPROACH TASK WITH V-SIDE GUSTS

The purpose of this section is to present a detailed examination of twelve $\psi \rightarrow \delta_r$ pilot closures representative of the approach task flown in the tests of Ref. 16. This includes an analytical determination of the rms heading response, σ_ψ, and the pilot's required control activity, $N_{\delta_r}\sigma_{\delta_r}$ to the simulated gust environment of the NRCC experiment.

The equation of motion and the values of the stability derivative used in this study correspond to those in Ref. 16. The NRCC experiments of interest here reduce to the single closed-loop rudder control of the heading disturbance. The control loop and the transfer functions have the form shown in Fig. 15. Values of the pertinent parameters and the stability derivatives for the approach conditions of Ref. 16 are

$$U_o = 50.6 \text{ ft/sec}$$

$$0.01 \leq N_v \leq 0.05 \quad \frac{1}{\text{ft-sec}}$$

$$0.1 \leq |N_r| \leq 10 \quad \frac{1}{\text{sec}}$$

The pilot model is assumed to be at most a gain, a time delay, and a lead,

$$Y_{p_\psi} = K_{p_\psi} e^{-.3s}\left(s + \frac{1}{T_{L_\psi}}\right)$$

The choice of the pilot model is not obvious for some of the cases examined, and a discussion of the considerations involved in choosing the pilot model is given later.

Based on the results of Ref. 16, the cases of Table B-X were chosen for study. Also listed are the dutch roll roots as well as the stability derivative of the helicopter transfer function.

TABLE B-X. DIRECTIONAL CASES

CASE NO.	N_v	$-N_r$	N_{δ_r} (opt)	ζ_d $\left(\frac{1}{T_{d_1}}\right)$	ω_d $\left(\frac{1}{T_{d_2}}\right)$
1	.01	0.1	.4	.07	.711
2	.01	1.0	.6	.703	.711
3	.01	2.0	.7	(.296)	(1.704)
4	.02	.1	.4	.050	1.01
5	.02	1.0	.7	.500	1.01
6	.02	2.0	.8	1.00	1.01
7	.035	1	.4	.377	1.33
8	.035	2	.6	.754	1.33
9	.035	5	.4	(.381)	(4.62)
10	.05	2	.7	.630	1.59
11	.05	5	1.5	(.576)	(4.42)
12	.05	10	2.3	(.26)	(9.74)

Parentheses indicate that the factors are real

Heading Closure

For the twelve cases the $\psi \rightarrow \delta_R$ loop was closed so that the crossover frequency was 2 rad/sec and the phase margin was at least 30°. A list of the pertinent closed-loop quantities is given in Table XI. Root locus and Bode plots for a selected group are given in Fig. B-12.

The importance of adequate yaw damping as a means of reducing pilot lead requirement is shown in Fig. B-12. In these cases the damping of the dutch roll mode is directly related to N_r, $N_r = -2\zeta_d\omega_d$. This is opposed to the pitch attitude case where the damping factor of the primary mode (phugoid) appears little changed by large changes in pitch damping, M_q, as shown in Fig. B-1. This need for greater damping in the directional case is further illustrated by closing the loop with a simple pilot model. As shown by Fig. B-12, increasing the loop gain primarily increases frequency rather than damping unless the pilot adapts leads of the order of 1 sec (which is accompanied by a rating degradation as large as 1 point). For the pitch case the opposite is true, i.e., increasing the loop gain goes primarily at first to increase the damping of the mode controlled (see Fig. B-2). This then is the reason why pilot rating is a strong function of damping in the directional case, and a weak function of damping in the pitch case.

The effect of N_v on the pilot closure is small except that an increase in N_v requires an increased N_r to maintain a given damping. Although increasing N_v has little direct effect on the closure, its primary influence is to increase the gust sensitivity as discussed in Section III-B-2.

Limitation of the Gust Simulation

In Ref. 16 the canned gust acted through N_v only. Due to this, one of the principal results of the NRCC study (i.e., for satisfactory pilot opinion at large values of N_v, N_r must be large) may be misleading. If a large portion of N_r was of aerodynamic origin, then the gust sensitivity of the simulator would be increased through N_r as well as N_v. Therefore, it was decided to briefly investigate this effect. In the Canadian simulation, only $N_v v_g$ was used as a gust input term. When N_r is of aerodynamic origin,

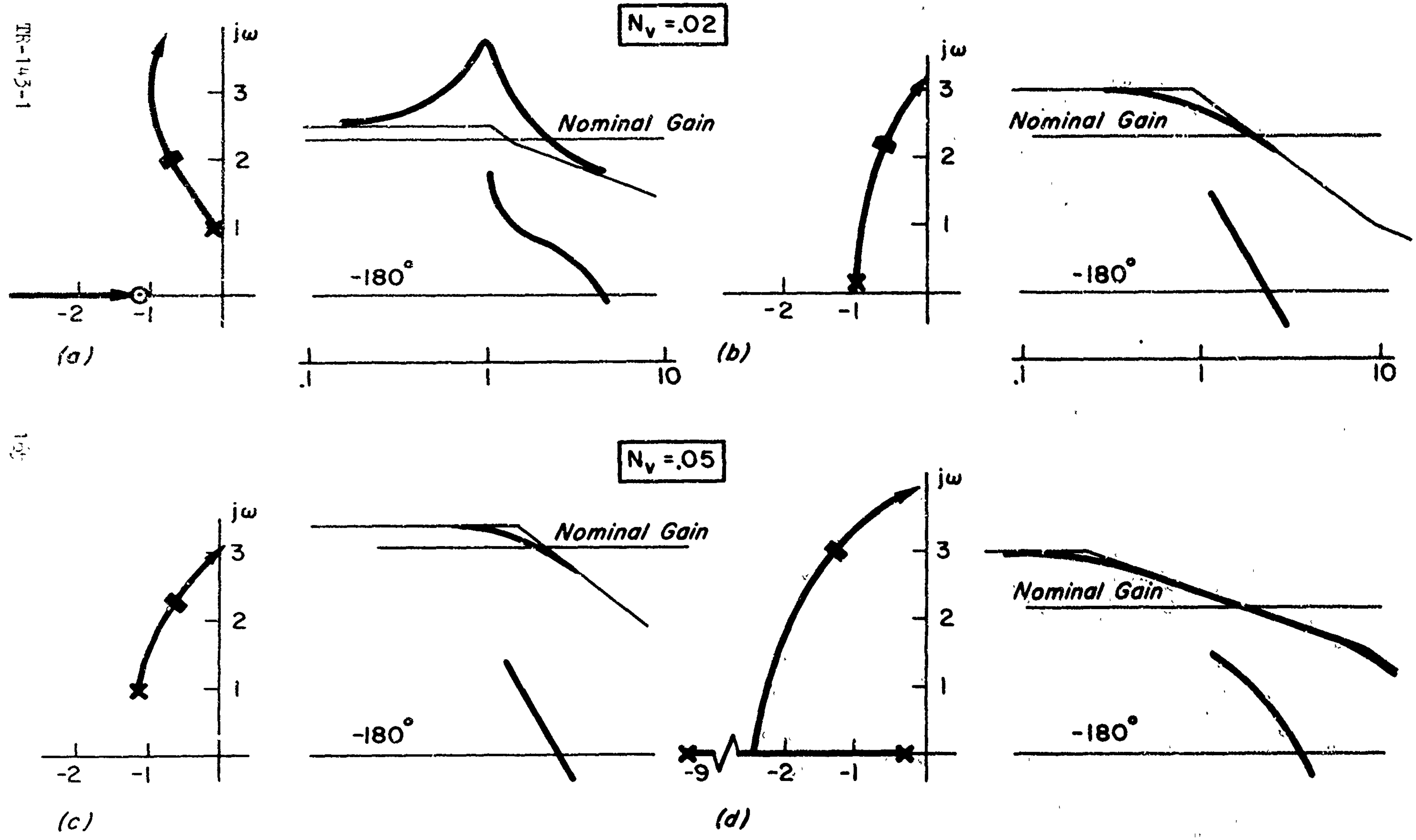


Figure B-12. Heading Closure for $\psi \rightarrow \delta_r$

a gust gradient term proportional to $N_r \dot{v}_g$ is important in determining directional gust responses. Thus the simulation in Ref. 16 contains the implication that N_v is totally aerodynamic and N_r is totally augmented. A brief investigation compared the σ_ψ/σ_{v_g} obtained by using Ref. 16 equations of motion with the σ_ψ/σ_{v_g} obtained by using modified equations in which $Y_v v_g \neq 0$ and $N_r s v_g/U_o \neq 0$. The gust transfer function for heading, ψ, becomes

$$\frac{\psi}{v_g} = \frac{-\frac{N_r}{U_o}\left(s - \frac{N_v U_o}{N_r} - Y_v\right)}{s^2 + 2(\zeta\omega)_d s + \omega_d^2}$$

For $\quad Y_v v_g = 0 \quad$ and $\quad \frac{N_r}{U_o} s\, v_g = 0 \quad$ (the case of Ref. 16)

$$\frac{\psi}{v_g} = \frac{N_v(s - Y_v)}{s\left[s^2 + 2(\zeta\omega)_d s + \omega_d^2\right]}$$

Thus, when the aerodynamic portion of $|N_r|$ is large (greater than 1.0), the pilot ratings in an actual gust would be somewhat worse than those reported in Ref. 16.

The Analytical Gust Model

The gust spectrum in Ref. 16 is equivalent to passing white noise of unity variance through the filter below

$$Y_{f_1} = \frac{(260)^{1/2}\left(\frac{1}{.0018}\right)s^2}{\left(\frac{s}{.0018} + 1\right)\left(\frac{s}{.253} + 1\right)\left(\frac{s}{.36} + 1\right)\left(\frac{s}{15} + 1\right)\left(\frac{s}{30} + 1\right)}$$

It was found that the simple gust model in Appendix B was qualitatively nearly identical to the above when ω_g was chosen as 0.5 rad/sec. The main discrepancies occur at frequencies below 0.3 rad/sec. Thus, for analytical purposes, the gust model chosen was the same as in the previous sections (see Appendix B-I).

RMS Responses to v-Gust

The determination of the rms ψ and $N_{\delta_\delta}\delta_r$ gust responses uses the same procedure as outlined in the hover case, Section B-I, of this Appendix. The ψ-response provides a measure of the pilot's performance in the approach task and $N_{\delta_r}\sigma_{\delta_r}$ provides a measure of both the control power and the control effort required of the pilot.

The rms values of ψ and $N_{\delta_r}\sigma_{\delta_r}$ were calculated for variations in pilot loop gains of Fig. B-12. The results for a medium N_v and the high N_v cases are shown in Fig. B-13. The trends also apply to the other eight cases. It can be seen from these figures that the nominal gains provide a reasonable compromise between the rms values of ψ and $N_{\delta_r}\delta_r$. The ψ-gain gives a near-minimum ψ-response for about the same control activity in each case.

Having selected the values of pilot gains, the effects on the rms responses of changes in the derivatives are determined for the twelve cases at the nominal gains. These values along with all pertinent closure information are given in Table B-XI. The rms values of the response are given as a ratio to the gust input. The correlation of these analytical results with the pilot rating of Ref. 16 is discussed under Section III-B-2 of the text.

Selection of a Pilot Model

The closures of this section appeared to have certain shortcomings when compared to the conventional adjustment rules for the human pilots describing function. The worst of these is the low dc gain over a long frequency region in the cases of low dutch roll damping. The dc gains in cases 4, 7, and 10 are all below 5 db. This indicates poor low frequency response to the primary forcing function (i.e., heading error).

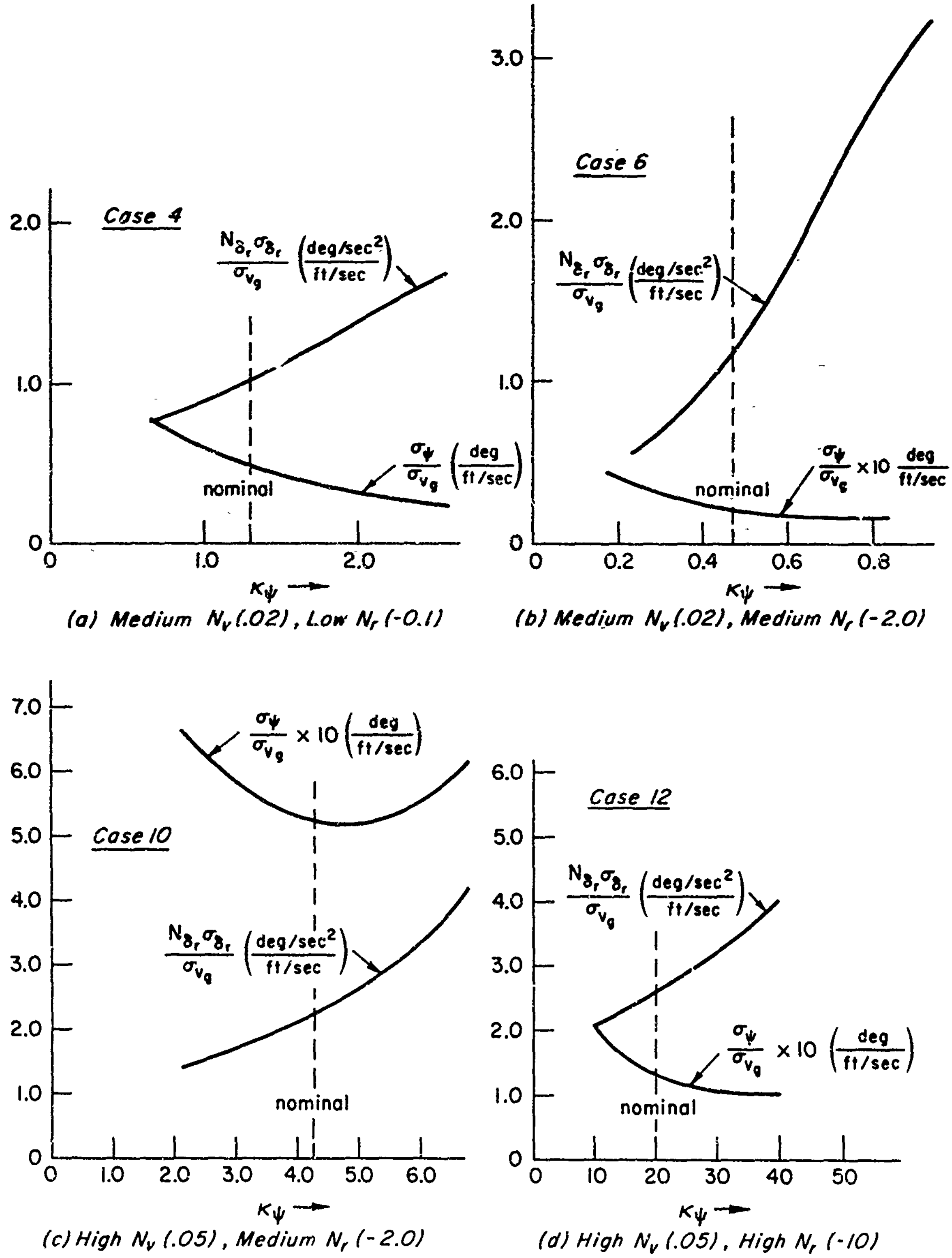


Figure B-13. Variations of Gust Responses with Pilot Gains

TABLE B-XI. SUMMARY OF $\psi \rightarrow \delta_R$ CLOSURE QUANTITIES AT NOMINAL GAINS

CASE NO.	DERIVATIVES		PHASE MARGIN	GAIN MARGIN	$\frac{1}{T_L}$	ROOT LOCUS GAIN K_ψ	BODE GAIN K_ψ	Δ' ROOTS* DUTCH ROLL		$\frac{\sigma_\psi}{\sigma_{v_g}}$	$\frac{N_{\delta_r}\sigma_{\delta_r}}{\sigma_{v_g}}$	N_{δ_r} (OPT)	POR AT OPTIMUM N_{δ_r}
	N_v	N_r	deg	db	1/sec	rad/sec	rad	ζ_d'	ω_d'	$\frac{\text{deg}}{\text{ft/sec}}$	$\frac{\text{deg/sec}^2}{\text{ft/sec}}$		
1	0.01	−0.1	30	9.3	1.11	1.53	3.36	0.384	2.02	0.261	0.582	0.4	6.5
2	0.01	−1.0	30	9.4	2.92	1.14	6.57	0.276	2.21	0.163	0.583	0.6	3.0
3	0.01	−2.0	30	7.2	7.31	0.7	10.1	0.236	2.35	0.115	0.595	0.7	2.9
4	0.02	−0.1	30	10.5	1.14	1.3	1.463	0.291	1.98	0.491	0.991	0.4	7.0
5	0.02	−1.0	30	9.6	3.45	0.9	3.06	0.224	2.19	0.317	1.05	0.7	5.0
6	0.02	−2.0	30	6.3	10.57	0.465	4.85	0.201	2.34	0.224	1.11	0.8	3.5
7	0.035	−1.0	30	8.7	4.91	0.566	1.57	0.163	2.18	0.541	1.56	0.4	8.0
8	0.035	−2.0	30	5.1	No lead	4.57†	2.58	0.157	2.33	0.389	1.78	0.6	6.0
9	0.035	−5.0	43.3	6.0	No lead	10.2†	5.76	0.280	2.82	0.180	1.84	1.4	3.0
10	0.05	−2.0	35.7	5.0	No lead	4.26†	1.68	0.149	2.38	0.525	2.24	0.7	7.5
11	0.05	−5.0	47.6	6.2	No lead	10.1†	3.98	0.280	2.91	0.245	2.47	1.5	4.5
12	0.05	−10.0	51.7	6.95	No lead	20.0†	7.9	0.388	3.29	0.132	2.63	2.3	2.5

*Crossover frequency, ω_c, for all cases is 2.0 rad/sec
†Units are $(\text{rad/sec})^2$

Also, in case 10, the pilot is crossing over in a K/s^2 region and not in the generally desired K/s region. To achieve the K/s crossover at the same 2.0 rad/sec, a sufficiently high pilot lead could be chosen. In case 10 this would put the pilot lead zero in the range 1.0 to 4.0 rad/sec. Any lead, however, will tend to worsen the already poor dc gain. Another alternative is the use of low frequency lag to improve dc gain and give a good K/s region at a lower crossover frequency and then sufficient lead to achieve good stability. When choosing a lower crossover frequency, it is desirable to keep ω_c somewhat larger than the gust break frequency of 0.5 rad/sec. In cases 1 and 4 any lag will force the crossover frequency well below 0.5. Thus, lag will not help these cases. In cases 7 and 10, it is conceivable that lag will improve the performance. Case 10 was chosen for study since it had no K/s region and would (due to higher ω_d) benefit the most from low frequency lag. The lag was chosen at 0.1 rad/sec and a lead term at 2.0 was added to give adequate gain margin. The results of this closure yielded a good K/s region in the vicinity of the crossover frequency (1.0 rad/sec); a phase margin of 53°, a gain margin of 7.5 db and a dc gain of 8.9 db. It would appear that this is a much better system than the "nominal" used for comparison purposes. When the rms values were computed, however, both σ_ψ and $N_{\delta_r}\sigma_{\delta_r}$ were larger than before and σ_ψ had increased considerably (i.e., from 0.53 to 0.8). Thus, the "better looking" closure with high dc gain gave much poorer gust response.

To check further into the lack of importance of dc gain on rms behavior, the case 10 was closed with a pilot lead at 1.0 rad/sec and no lag. Here the dc gain was very low (0.75) but σ_ψ was close to the original value while $N_{\delta_r}\sigma_{\delta_r}$ was greatly reduced (2.6 to 1.4). The summary of closed-loop parameters and rms responses for the three pilot models is given in Table B-XII.

Since the pilot can do so well with large lead and poor dc gain, one might be tempted to conclude that the lead only case would most closely represent the pilot model. The pilot, however, is first concerned with good low frequency tracking of the primary signal, the heading error. Thus, the pilot will attempt to keep his crossover frequency high enough to keep the rms responses low but use as little lead as possible to achieve a good rating.

TABLE B-XII. SUMMARY OF THE CASE 10 STUDY

CONDITION	ω_c	PHASE MARGIN	GAIN MARGIN	LEAD $\frac{1}{T_L}$	LAG $\frac{1}{T_I}$	GAINS ROOT LOCUS κ_ψ	GAINS D.C. K_ψ	DUTCH ROLL ROOTS ζ_d'	DUTCH ROLL ROOTS ω_d'	$\frac{\sigma_\psi}{\sigma_{v_g}}$	$\frac{N_{\delta_r}\sigma_{\delta_r}}{\sigma_{v_g}}$
	$\frac{RAD}{SEC}$	DEG	db	$\frac{RAD}{SEC}$	$\frac{RAD}{SEC}$		RAD			$\frac{DEG}{FT/SEC}$	$\frac{DEG/SEC^2}{FT/SEC}$
(a) ORIGINAL NO LEAD NO LAG	2.0	35.7	5	—	—	$\left(\frac{rad}{sec}\right)^2$ 4.26	1.68	.149	2.38	.525	2.24
(b) LAG-LEAD	1.0	52.5	7.5	2.0	0.1	$\left(\frac{rad}{sec}\right)^2$ 1.13	8.94	.448	1.43	.797	2.59
(c) LEAD ONLY	2.0	99	10	1.0	—	$\frac{rad}{sec}$ 1.91	0.755	.543	4.5	.571	1.42

APPENDIX C

SURVEY TRIP NOTES

on

HELICOPTER HANDLING QUALITIES MIL SPEC 8501A

FT. RUCKER. 11/13/64

Victor Holloway	WO 1
James H. Nichols	Major
Don C. Jones	Civilian
Edwin N. Clay	Major
Charles E. Arnold	Civilian
Vernon R. Evans	Captain
O. A. Buettner	Civilian
Womack	WO

NASA-LANGLEY. 11/12/64

J. Reeder	
R. Tapscott	
J. Kelly	
R. Piper	TRECOM

VERTOL. 11/11/64

Jim Smith	Hugh McCafferty
John Kannon	E. Diamond
Frank Duke, Pilot Section	P. Sheridan
Bob Gangwish, Pilot Section	B. Blake
Bob Stroub	K. Landis
J. W. Berkeley	

PATUXENT. . . . . 11/10/64 (Rotary Wing Branch, Flight Test Division)

Lt. D. A. Beil	NATC Flight Test
Capt. P. L. James	NATC Flight Test
W. R. Powers	NATC Flight Test
R. C. Lamond	NATC Flight Test
Lt. D. F. Mayers	NATC Flight Test
Ben Stein	BuWeps

SIKORSKY. 11/9/64

E. Carter	D. Jenny	W. King
J. Needen	D. Cooper	R. Perrone
R. Baker	J. Clark	P. D'Estillio
T. Kaplita	D. Gardner	

HUGHES. 10/29/64

K. Amer

C. Savant

LOCKHEED. 10/28/64

Ford Johnson

Carl Friend

SYSTEMS TECHNOLOGY, INC. REPRESENTATIVES

I. L. Ashkenas

D. Graham (Replaced Mr. Ashkenas at Sikorsky)

R. P. Walton

SPEC ITEMS	COMMENTS
GENERAL (NONSPEC)	**FT. RUCKER** Don't yet have <u>military</u> type of vehicle, i.e., <u>carrier</u>. Off-the-shelf not so good. Want something like a flying crane having a pod attachment for every mission. Want utility--not concerned with handling qualities, except for lousy <u>seats</u>, and large range of trim attitudes for fuselage. **HUGHES** Spec should be written only when there is a demonstrated need, e.g., for $\partial\delta/\partial u$ to have stable slope takes lots of effort, but it's only in helicopter spec as a holdover from the airplane spec. <u>Not required for hover</u>. No evidence to show that it does improve hover characteristics.
3.1.2 Definition of Normal Service Loading	**PATUXENT** Normal service loadings—spend a lot of time wondering what this is.
3.2.1 / 3.3.4 Control Power in Unaccelerated Flight for Disturbances 3.3.6 Directional Control Margin for 35 Kt Wind from Critical Direction	**FT. RUCKER** Don't think any present helicopter will meet <u>present</u> crosswind <u>turn</u> requirement but spec item OK, but not sure would want to sacrifice power to achieve. **PATUXENT** Maximum CP margin not specific—direction, c.g., G.W.
3.2.13 / 3.3.5 / 3.3.18 Control Power in Terms of Response to Unit and Maximum Control Step Input	Control power test OK. Tandems do not have precise <u>directional</u> control, but meet control power requirement—large inertia, low damping. For <u>utility</u>, most maneuvers limited to 30° bank. For <u>weapons</u>, bank angles can go as high as 90° In <u>attack</u>, time-to-bank/time-to-maneuver $<$ constant. Never have encountered longitudinal or lateral control power deficiency. Do limit controls because of accelerations they feel.

SPEC ITEMS		COMMENTS
3.2.1 3.3.4 3.3.6 3.2.13 3.3.5 3.3.18	Continued	

SIKORSKY

Control sensitivity (M_δ) determined by CP_{max} and cockpit size: $= M_\delta \delta_{max}/\delta_{max}$.

If meet spec for trim in high speed and rearward flight, then M_δ may not be satisfactory.

Requirement for gain changers exists as speed range becomes greater, but Sikorsky opposed to pinning down M_δ.

Sikorsky basically accepts control power requirements.

M_δ available for hover makes machines more agile than they need to be.

S-61 can fly same envelope as smaller machine.

As to $M_\delta \delta_{max}$ — How is it measured or how can demonstration be made of the 10% requirement?

Claim spec requires too much N_δ.

VERTOL

For precise <u>heading</u> control they are presently criticized for low sensitivity.

Their present yaw rate capability is less than MIL-H-8501A; CH-46, 47 designed to -8501.

Sikorsky restricts to 360°/15 sec—too slow. They gave customer 360°/10sec, which felt was very adequate.

Rapid turning has lead to pitching motions—can lead to loss of pitch control.

Not important for transports, but could be very important for Escort fighter.

HUGHES

Critical roll maneuver is probably obstacle avoidance.

Adequate CP first prerequisite, then gradient (M_δ, L_δ, etc.).

Maximum $L_\delta \delta$ — hovering in 30 kt crosswind and asymmetric loading more critical than roll in 1 sec.

Upsetting moment about all axes is proportional to control power, as is damping, e.g., offset hinges increase the upsetting moments in the same proportion as the hinge increases the control power and damping.

SPEC ITEMS	COMMENTS

3.2.1 Continued
3.3.4
3.3.6
3.2.13
3.3.5
3.3.18

LOCKHEED

Gust sensitivity (M_u) increases with disk loading (higher collective) and disk loading increases with size.

Step side wind of 35 kt for control more critical than any maneuver requirement.

3.2.2 Steadiness in Hover
3.3.3

NASA-LANGLEY

For hovering over spot, Canadians show gust effects. Their tasks moving near ground—coming up to a spot rather than hovering over it — don't show gust effects.

Want to limit M_u to low value.

Think initial rotor tilt is helpful in translating as opposed to jet VTOL.

Even without gusts, always disturbance due to hovering, e.g., recirculation disturbances, etc.

3.2.4 Longitudinal and
3.3.11 Lateral Force Gradients

3.2.6 Longitudinal and
3.3.12 Lateral Trim Forces

3.2.7 Breakout Forces
3.3.13

3.2.10 Longitudinal Stick Stability (Position and Force)

VERTOL

Stick forces. Breakout and gradient—recognize valve friction—artificial breakout force around 1 lb, including valve friction.

They like the forces in Table II for hover, except for rudder. They equate desirable gradients with displacements required for holding—and frequency of occurrence.

Around 60 kt, basic tandem has zero stick gradient. Therefore, had poor trimmability (S.F.).

Level flight stick stability not important ($\gamma = 0$)

As to stick forces versus stick position, they are stick force men, not stick position

PATUXENT

Stick forces in Table II are too low in all cases.

Level flight trim versus airspeed—want stable slope to indicate speed change.

Also say stick force gradient good on approach.

SPEC ITEMS	COMMENTS

3.2.4 Continued
3.3.11
3.2.6
3.3.12
3.2.7
3.3.13
3.2.10

SIKORSKY

Control breakout forces, especially directional, too low.

Position stick gradients—are they necessary?

Sikorsky does not always meet 3.2.10 at 50 kt, especially with large tail sizes.

Sikorsky says LA Airways flies with no feel in VFR. They don't care about force gradient and they may not care about position gradient (constant collective, constant throttle).

LOCKHEED

Trim requirement:

7 lb maximum lateral stick force

2 to 2.5 lb maximum longitudinal for Lockheed helicopter

3.2.11 Longitudinal Dynamic Stability

VERTOL

They prefer pulsing elevator and looking at ω and ζ.

With SAS they aim for $\omega \doteq 3$ rad, $\zeta_{sp} \doteq 0.7$–0.8.

SIKORSKY

Oscillatory requirements—basically OK.

IFR should extend over entire frequency range and numbers should be reviewed from closed-loop basis.

Blade stall is limit of linear theory because large motions (1" pull and hold) give nonlinear response, especially in high speed machines.

3.2.11.1 Longitudinal Maneuver Stability

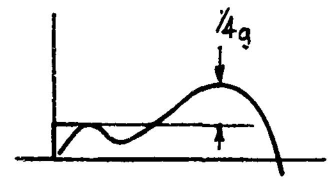
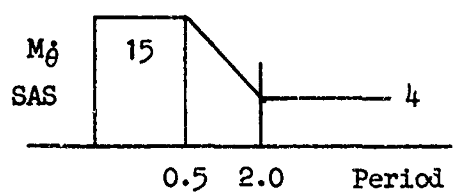
NASA

The "concave" downward requirement was substitute for minimum maneuver margin. Originally they could not hold V to do maneuver for maneuver margin.

PATUXENT

Maneuver requirements should be identical with mission, i.e., for attack the "concave" downward requirement should be similar to airplane spec.

Basic criteria OK — not sure about numbers (2 sec).

SPEC ITEMS	COMMENTS
3.2.11.1 Continued	**SIKORSKY** Maneuver margin is perhaps better, or the acceleration trim gradient. **HUGHES** The "concave" downward requirement still considered good. **LOCKHEED** On "concave" downward—aperiodic divergence may come after first cycle of phugoid—a poor indicator.
3.2.11.2 Normal Acceleration Stay Within 1/4g After 1/2 Second Pulse	**SIKORSKY**  Believe exceedance of 1/4g should be measured as shown.
3.2.14 3.3.19 Damping Required to Insure Satisfactory Initial Response and to Minimize the Effect of Disturbances	**NASA** $T_R < 2$, $-1/N_r > 4$, for minimum right now, but Reeder thinks these 'll come down. **VERTOL** $1/M_q$, $1/L_p$ close to 0.8 for basic aircraft, but no good because M_α is positive—when add SAS to pitch, increase effective M_q.  **PATUXENT** Damping criteria difficult to measure, since an attitude change starts aircraft to translate.

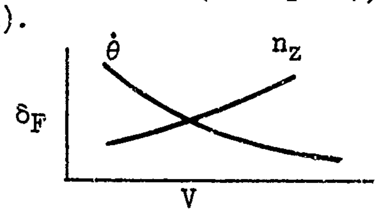
SPEC ITEMS	COMMENTS
3.2.14 Continued 3.3.19	**SIKORSKY** For Sikorsky ships, M_q is approximately constant with size: = 0.5 basic, = 1.8 with ASE. N_r — Too much required. OK with ASE. Might meet with nonlinear effects. **LOCKHEED** Rigid rotor. $L_p \doteq 6$, $M_q \doteq 1.8$.
3.3.5 Directional Yaw Response to Unit and Maximum (Control Power) Control Step	**HUGHES** Yaw acceleration capability — pilot location forward of c.g., therefore wouldn't subject self to high linear accelerations; therefore expect size effect. However would expect no size effect on the ability to achieve a new heading in short time; and on the equivalent situation in roll — to move over laterally in short time.
3.3.9 Positive Directional Stability, N_β, and Dihedral, L_β — δ_r, δ_a versus β Linear —10% Control Margin for Gust in Lateral Control	**FT. RUCKER** Pedals no real use in most flying, except for taxiing, hovering, takeoff, and landing. Would like to fly without pedals most of the time—over 50 ft altitude. Most pilots don't really learn to use pedals.
3.3.9.1 For Speeds Above 50 Kt or 1/2 V_{max}, Complete turns Without Pedals and Without Rolling velocity Reversals 3.6.2 Extension of Above Specs to Include Control Force Stability	**VERTOL** 15° β requirement at V_{max} designs their rudder control. There is no operating requirements—why is it in? 45° at 60 kt is too high for visibility — e.g., plane guard task on carrier recoveries. Question as to whether demonstrate with or without autopilot — then what about stick-fixed stability? 10% in control power is better than old spec, which called for 10% in position. Maximum attainable (ambiguous)— neutral to extreme rather than extreme to extreme; which is correct interpretation? **PATUXENT** Linearity may not be desirable. Also, 15 lb too low —see conventional airplane spec.

SPEC ITEMS		COMMENTS
3.6.2	Continued	**SIKORSKY** Pedal force proportional to sideslip very questionable with heading retention. Artificial force may be desirable (to keep synch switch closed).
3.3.17	Lateral Trim Changes Shall Not Be Excessive With Changes in Power or Collective Stick	**PATUXENT** Rudder pedals required for transition into autorotation in single rotor helicopter — should be reduced. Throttle cross-tie — very troublesome.
3.5.5	Safe Transition Into Autorotation	**PATUXENT** Delay time for engine failure: 2nd failure — shorter time delay 1st failure — 2 sec seems reasonable 1 sec for one recip. engine Up to V_{max} recovery from engine failure. h-V diagrams usually drawn for level flight—wants climb with takeoff power. **SIKORSKY** As to delay times from unalerted testing — 0.5 sec appears more realistic than the 2 sec specified.
3.5.7	Autorotative Landings at Speeds Less than 15 Kt	**PATUXENT** Have trouble with 3.5.7. **SIKORSKY** Autorotative — 15 kt, one engine out, perhaps, but Sikorsky finds this difficult to achieve in power-off landings.
3.5.9	Use of SAS or ASE to Meet Spec Requirements	**FT. RUCKER** ASE lots of trouble in maintenance. Fatigue without ASE, but not mandatory for flight. Don't need ASE for all jobs. Basic machine — can't fly hands off. One hour in helicopter is equivalent to 3 hr in airplane.

GENERAL TOPICS	COMMENTS
CONTROL AND MANEUVER REQUIREMENTS FOR ARMED HELICOPTERS	**FT. RUCKER** 30°/sec to 50°/sec per inch sounds high to them. Also question of high speed, i.e., how limit speed to retain control? Flexible turrets will be used for area fire. Consensus seems to be that Bell (Edenborough's 20th AHS Forum article) rates are too high!
"HARMONY"	**VERTOL** Complain about weak roll control power versus high pitch — never use maximum roll control. Harmony can be expressed in a number of ways, i.e., do you harmonize forces, displacement or maximum control power. More study needed in this area.
LATERAL-DIRECTIONAL DYNAMIC STABILITY	**VERTOL** No spec on lateral-directional dynamics for VFR. If supply specified N_r and have static directional stability, could have horrible characteristics
NASA TN D-2477 TASK	**NASA** Task--All axes under hood. Level attitude during S-maneuver most important. β low and high Dutch roll damping. High N_β rough on passengers up through moderate turbulence. Most helicopters don't have adverse yaw (N_p). Do on rigid rotor. 8–10 sec to make maneuver back to course. Therefore if Dutch roll period in this region, pilots want more damping. Damping correlations ($\zeta = N_r/2\sqrt{N_\beta}$) only for $N_\beta > 0.4$ for TN D-2477.

GENERAL TOPICS	COMMENTS
PILOT-INDUCED OSCILLATIONS	**NASA** Two types of PIO's have occurred: Long period, large amplitude, full throw due to low damping and control power High frequency, high sensitivity SC1, P1127--increased CP and sensitivity; VZ-4. Low control power, low damping in hover. SC-1, VZ-4, P1127 cited; last improved with increased CP and sensitivity. **SIKORSKY** Flying crane PIO due to cable and 1/rev of rotor. Cured by vibration isolator on cable.
PROBLEMS OF TANDEM	**VERTOL** Two basic problems-- $-M_\alpha$, $N_\beta < 0$, actually unstable at cruise except at forward c.g. (for M_α). c.g. limit set by high speed and rotor loading rather than stability or control. L_β is high--to augment N_β use static pressure in nose--Δp transducer for $V > 60$ kt. Can't use a_y to get <u>static</u> stability in sense of δ_r or φ <u>versus</u> β. $N_\beta \doteq 0.3$ about as low as want to go — difficult to control for low values.

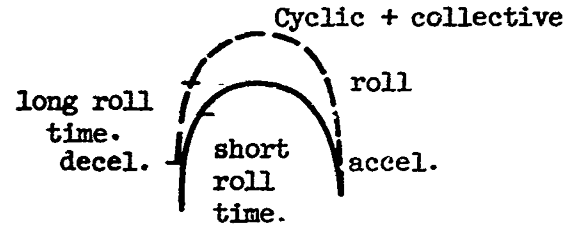
GENERAL TOPICS	COMMENTS
STICK FORCE PER G - contd	**VERTOL** Doubt effectiveness at high speeds due to blade stall limitations. **PAX** SF/g difficult to get at low speed — of opinion that one cannot get much below 50 kt; OK above minimum power. SF gradient for hover desirable to indicate forward velocity — a good airspeed indicator not available. **HUGHES** Kaman has a hydraulic bobweight — very successful at 15 lb/g; first service application he knows of. Tried bobweights at NASA-Langley at about 60 kt — pilots unenthusiastic. Ordinarily do not get force signal until n_z is achieved. Low spring gradient needed for hover. Most helicopters have power actuators. If not, stick forces are unpredictable. **LOCKHEED** SF/g is a good substitute for concave downward criteria.

GENERAL TOPICS	COMMENTS
SIZE EFFECTS - contd	**LOCKHEED**
	From dimensional analysis, $I \propto L^5$, $W \propto L^3$, $W/A \propto L$.
	Desire large D/I to prevent gust upsets because of small size (referring to rigid rotor damping). Also leads to higher rotor (Th) because of ground clearance.
SIZE VERSUS MISSION	**FT. RUCKER**
	Suggest different spec for different missions.
	NASA
	Size effect was expedient at time spec was written.
	Minimums are compromises.
STICK FORCE GRADIENT	**LOCKHEED**
	Wants a stick force gradient as opposed to none for light damping.
STICK FORCE PER G	**FT. RUCKER**
	Kaman bobweight liked.
	But must have light forces around trim because in gusty air, pilot is fighting the rotor flexing.
	Says SF/g good idea.
	NASA-LANGLEY
	Do want maneuvering force — are planning work in this area to get stick force (δ_F) proportional to $\dot{\theta}$ (low speed), g (high speed).
	
	Also important cue as well as limiting — used 14 lb/g damper.
	Force cue desired — AGARD spec.

GENERAL TOPICS	COMMENTS
ROLL MANEUVERABILITY	**LOCKHEED**

Desires fast roll to keep flight path shorter.

High rotor inertia provides additional power in turn.



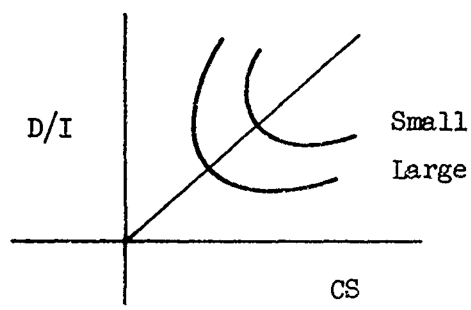
For obstacle avoidance -- just slow down rather than increase roll capability.

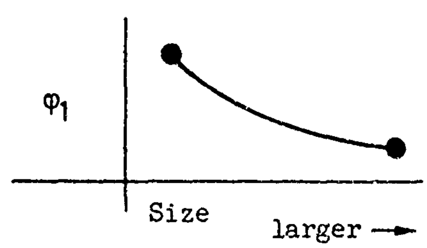
On Lockheed, $p_o = 35^\circ$ to 45°/sec.

Required to avoid obstruction -- get rotor around.

SIZE EFFECTS	**NASA**

Studying size effect by plotting roll response versus size.





$\sqrt[3]{W + 1000}$ is <u>really</u> supposed to be <u>size</u> effect rather than weight.

Makes point about optimum T_R versus minimum T_R. These may indeed vary with size.

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13. ABSTRACT

This report is directed at analysis and review of the current military helicopter flying qualities specification, MIL-H-8501A, and of the relevant published literature. The analytical approach rests primarily on servo-analysis of closed-loop piloting tasks and secondarily on open-loop response considerations, and such analysis and considerations are used as the basis for correlating and "explaining" the available data. This process delineates those flying qualities criteria (either in the specification or proposed in the literature) which are valid, and establishes bases for additional experiments in inadequately explored critical areas. The results are specifically applied to an itemized critique of MIL-H-8501A (excluding autoration, miscellaneous, instrument flight, and vibration characteristics).

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14. KEY WORDS	LINK A		LINK B		LINK C	
	ROLE	WT	ROLE	WT	ROLE	WT
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